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VOLUME I

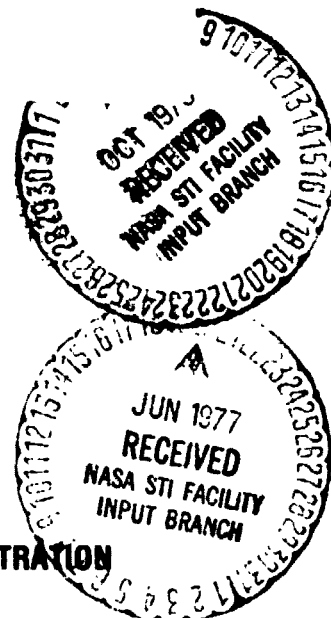
Tasks I and II Final Report

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16 Abstract A structural design study was made, based on a 1975 level of technology, to assess the relative merits of structural concepts and materials for an advanced supersonic transport cruising at Mach 2.7. Preliminary studies were made to insure compliance of the configuration with general design criteria, integrate the propulsion system with the airframe, select structural concepts and materials, and define an efficient structural arrangement. An advanced computerized structural design system was used, in conjunction with a relatively large, complex finite element model, for detailed analysis and sizing of structural members to satisfy strength and flutter criteria. A baseline aircraft design was developed for assessment of current technology and for use in future studies of aerostructural trades, and application of advanced technology. Criteria, analysis methods, and results are presented. An appraisal of the effect of the use of the computerized structural design system on design methods and recommendations concerning further development of design tools, development of materials and structural concepts, and research on basic technology are also presented.			
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SECTION I

CONFIGURATION DEVELOPMENT

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SYMBOLS AND ABBREVIATIONS

AB	Abreast
A B	Afterburner
ALT	Altitude
A P	Airplane
APP	Approach
APU	Auxiliary power unit
AR	Aspect ratio
AST	Advanced supersonic transport
ATAT	Advanced technology afterburning turbojet
a_x	Longitudinal acceleration
a_y	Lateral acceleration
BL	Buttock line
Btu	British thermal unit
b	Wing span
b 2	One half span
C_D	Drag coefficient
CG.cg	Center of gravity
C_L	Lift coefficient
C_{ℓ}	Rolling moment coefficient
C_{LAPP}	Lift coefficient for landing approach
C_{LMAX}	Maximum value of lift coefficient
$C_{LMAXDEM}$	Maximum value of lift coefficient to be demonstrated
C_M, C_m	Pitching moment coefficient

Symbols (Continued)

C_{M_0}	Pitching moment coefficient at zero angle of attack
$C_{M_{TAIL}}$	Pitching moment coefficient due to horizontal tail
C_n	Yawing moment coefficient
$C_{N_{MAX_{DEM}}}$	Maximum demonstrated normal force coefficient, $\frac{\text{Normal force}}{qS}$
$C_{n\beta}$	Static directional stability derivative
c_R	Root chord
c_T	Tip chord
c_W	Wing chord
$\bar{c}$	Mean aerodynamic chord
$\bar{c}_{REF}$	Reference chord
$c/4$	One quarter chord (reference point)
D	Drag
DB, db, dB	Decibels
$Deg. ^\circ$	Degrees
dia	Diameter
ECS	Environmental control system
$EPNdB$	Effective perceived noise decibels
F	Fahrenheit
FAR	Federal Aviation Regulation
F_N	Engine net thrust

Symbols (Continued)

FT.ft	Feet
G	Control elements in a block diagram
g	Gravitational acceleration
hr	Hours
HSAS	Hardened stability augmentation system
in.	inches
K	Kelvin
KEAS	Knots-equivalent airspeed
kg	Kilograms
km	Kilometers
kts	Knots
LB.lb.lbs	Pounds
lbf	Pound force
L E.	Leading edge
L D	Lift over drag
Ldg	Landing
L D _{MAX}	Maximum value of lift over drag
l_H	Horizontal tail arm length
l_V	Vertical tail arm length
(L W)	Load factor (lift over weight ratio)
M	Mach number
m	Meters

Symbols (Continued)

MAC	Mean aerodynamic chord
M_{MO}	Maximum operational mach number
mtd	Mounted
MTW	Maximum taxi weight
M_{δ}	Pitching moment derivative with respect to control surface deflection
N	Newtons
N.Mi., n mi	Nautical miles
n_z	Normal acceleration
OEW	Operational empty weight
p	Roll rate
PPD	Prototype Point Design
PR	Percent Radius
psf	Pounds per square foot
PT	Prototype
q	Dynamic pressure
qSb	(Dynamic pressure) (Area) (Span)
q_0	Freestream incompressible dynamic pressure
R	Rankin
r	Yaw rate
rad	radians
RFP	Request for proposal
S	Laplace transform variable

Symbols (Continued)

s,sec,SEC	Seconds
S & C	Stability and control
SFC	Specific fuel consumption
S _H	Horizontal tail area
SLS	Sea level static
SOB	Side of body
SQ,sq	Square
S _{ref}	Reference area (usually wing area)
S.S	Sound suppressor
SST	Supersonic transport
Std	Standard
S _V	Vertical tail area
S _W	Wing area
T	Temperature, Time
t	Thickness
t/c	Thickness ratio (thickness over chord)
TE,TE.	Trailing edge
T.O.	Take-off
T.R	Thrust reverser
T _{EXIT}	Engine nozzle exit total temperature
t/2c	One half thickness ratio
V,V _{TR}	True Velocity (speed)

Symbols (Continued)

V_{APP}	Approach speed
V_{BAL}	Speed for engine out yawing moment
V_C	Calibrated cruise speed
V_E, V_C	Calibrated equivalent speed
$V_{MIN DEM}$	Minimum demonstrated speed
V_R	Takeoff rotational speed
V_{MO}	Maximum operational speed
V_1	Takeoff safety speed
$\bar{V}$	Tail volume coefficient
$\bar{V}_H$	Horizontal tail volume coefficient $= \frac{S_H l_H}{S_W l_{REF}}$
$\bar{V}_{HMIN}$	Minimum value of horizontal tail volume coefficient
$\bar{V}_V$	Vertical tail volume coefficient $= \frac{S_V l_V}{S_W l_{REF}}$
W	Watts
w	with
WL	Waterline
w/o	Without
x/c	Longitudinal (fore and aft) distance over chord
z/c	Vertical distance over chord
ZFW	Zero fuel weight
$2Y/b$	Fraction of wing semispan

Symbols (Continued)

α	Angle of attack
α_{LOF}	Liftoff angle
α_{WRP}	Rotation angle in relation to wing reference plane.
$\dot{\alpha}$	Time rate of change of α
β	Sideslip angle
Δ	Delta (incremental)
δ	Deflection angle (usually control surface)
δ_a	Aileron deflection
δ_e	Elevator deflection
δ_F	Flap deflection
δ_H	Stabilizer deflection
δ_r	Rudder deflection
δ_{SD}	Spoiler deflection
η_{IB}	Inboard engine location in fraction of semispan
η_{OB}	Outboard engine location in fraction of semispan
θ	Pitch angle
$\dot{\theta}$	Pitch rate of change
$\ddot{\theta}_{MAX}$	Maximum pitch acceleration
Λ	Sweep angle
λ	Taper ratio
μ	Forward velocity increment
ϕ	Bank angle

Symbols (Concluded)

ϕ_t	Bank angle achievable in time t
$\phi_{2.5}$	Bank angle achievable in 2.5 sec.
ϕ_7	Bank angle achievable in 7 sec.
$\ddot{\phi}$	Rolling acceleration
ψ	Heading angle
ω	Vertical velocity increment

SECTION 1

CONFIGURATION DEVELOPMENT

INTRODUCTION

The airplane configuration on which the structural analyses were conducted is an arrow wing concept most representative of a 1975 technology level. The NASA request for proposal (RFP) configuration, figure 1-1 designated as model 969-510 has been reviewed and modified to ensure that it incorporated the latest technology and satisfied basic flight requirements.

The configuration requirements were for a cruise Mach number of 2.7 with a maximum taxi gross weight (MTW) 340,194 kilograms (750,000 lb), a payload of 22,222 kilograms (49,000 lb) which represents 234 passengers in tourist accommodations. The range objective was set at 7778 kilometers (4200 N.Mi.) assuming a maximum landing weight of 190,509 kilograms (420,000 lb). The achievement of the above landing weight must finally depend upon the operating empty weight of the configuration and the reserve fuel requirements. The airplane center of gravity limits must be consistent with the stability and control criteria and the configuration was balanced to keep the operational center of gravity within the allowable limits. The propulsion definition was established by the contractor and was a composite selection of engine characteristics to account for an advanced technology engine cycle to be defined by the engine manufacturers during the NASA AST Engine Cycle Studies. The propulsion pods were sized to house the engine using the NASA Ames translating center body type intake. Nonoptimum features of the RFP configuration were altered so that they complied with the above requirements. Two alternate configurations using the modified RFP planform were drawn and evaluated. These were designated 969-512 and 969-513 and were chosen to allow for a comparison of the impact of engine location on the arrow wing planform. The 969-512 featured an outboard engine location retaining the lift advantage of the large inboard flap. The 969-513 featured an inboard location of the engines which reduced the impact on the control system requirements to handle outboard engine failures. The 969-513 configuration with the pods located inboard was heavier, leading to the selection of 969-512 with the engines located outboard on the wing trailing edge similar to the NASA RFP configuration. The chordwise location, diverter height, intake and nozzle inclination of the engine nacelles were selected to minimize drag and enhance the inlet performance. The 969-512 configuration was cycled once again to incorporate an increase in maximum flap angle and was designated the 969-512B.

Several alternate engine locations were identified on the 969-512B for the purpose of estimating structural and performance sensitivity. Subsequent to the configuration selection the fuel tank arrangement was identified. The arrangement was selected to accommodate the fuel management scheme as well as to minimize the thermal management requirements.

REVIEW OF NASA "REQUEST FOR PROPOSAL" (RFP) CONFIGURATION (MODEL 969-510)

The configuration defined by the NASA for use in conducting the structural analysis under NASA Contract NAS1-12287 is designated Model 969-510 by the contractor and is shown in figure 1-1. This configuration is similar to the Model 969-336C discussed in reference 1-1, except for modifications to the wing leading edge sweep angle to provide additional wing chord and box depth on the outboard wing, and an outward shift of the engines to provide for a larger inboard flap to improve low speed lift. In addition to the above changes, the canard and "dema" leading edge flap systems were removed.

The 969-510 configuration has been reviewed for completeness of definition and for features which could be altered to improve its flight characteristics or performance. Modification to the 969-510 configuration resulting from this review consisted of the following:

1. A reduction in wing tip leading edge sweep angle from 1.127 radians (64.6 degrees) to 1.047 radians (60.0 degrees). This provides an increase in the wing tip lift capability and improves the pitching moment characteristics of the arrow wing.
2. The planform of the horizontal stabilizer was changed to improve its effectivity and the area increased to satisfy nose down control (stall recovery) criteria (see lateral-directional stability and control criteria discussed later on in section 1).
3. The all movable vertical tail was changed to a fixed vertical with segmented rudder due to the increase in lateral-directional control requirements necessary to satisfy engine-failure control criteria (see longitudinal stability and control criteria discussed later on in section 1).
4. The airplane zero fuel weight center of gravity was repositioned within acceptable limits by shifting the body on the wing.
5. The wing thickness was increased over the wheel well area to accommodate the landing gear without requiring an out-of-contour fairing on the upper surface.
6. A nose down twist of .026 radian (1.5 degrees) was added to the wing tip to optimize the trim drag at cruise.
7. An engine representative of a second generation supersonic propulsion system was defined by combining the performance characteristics of a GE4 J6G with the geometry of a GE4 J6H2 sized for 287 kg/s (633 lb/sec) airflow and the weight based on a scaled GE21 J2B1 (see weights discussed later on in section 1).
8. The fuel tank arrangement was altered to locate fuel in the aft body and in the deeper wing sections to alleviate the fuel heating during supersonic cruise.

ALTERNATE CONFIGURATIONS (MODELS 969-512 & 513)

Two configurations incorporating the aforementioned modifications are the basis for studies of the impact of engine location on airplane weight and performance. One configuration shown in figure 1-2, designated 969-512, has the engines located on the trailing edge at 32 and 55 percent span, identical to the 969-510 configuration in the F-15. The second configuration shown in figure 1-3, designated 969-513, has the engines located inboard at 11.6 and 38 percent span. Both configurations are drawn with the same engine definition and the same maximum taxi gross weight. Takeoff field length objectives of 3780 meters (12,400 ft) and flight control criteria were also satisfied by each of the above configurations.

ENGINE SELECTION

The engine defined for the above configurations represents an advanced technology engine and is designated ATAT-1. It was originally anticipated that engine data supplied by NASA would be used for this contract; however, these data were not available and it was deemed desirable to proceed with a specific engine as requested by the contractor in reference 1-2. This engine is the same engine selected for the final A&T Task III study (Contract NAS1-11938) and is described below.

ENGINE DEFINITION

Figure 1-4 shows the overall pod geometry of the ATAT-1, and the source or basis for geometry, weight, performance and noise.

Basically, the synthesis consisted of building on the GE4 J6G afterburning turbojet, by modifying its geometry, weight, performance and noise, based on data for more advanced technology designs. For example, when an annular, plug-type, nozzle is substituted for the original two-stage ejector nozzle, the assumed weight, geometry, afterburning temperature limit, and suppressor characteristics were all consistent with the new nozzle design.

GEOMETRY

The geometry is based on the GE4 J6H2 engine, modified for the addition of an afterburner, and the contractor's N-5 axisymmetric intake. The J6H2 engine and nozzle geometry is well defined, and representative of an advanced technology design. After scaling to the selected size, the tailpipe length and diameter are increased to reflect the addition of an afterburner. A 0.35 radian (2 degrees) bend in the intake, and a 0.70 radian (4 degrees) bend in the tailpipe upstream of the afterburner flame holder are incorporated to reflect the integration requirements of the pod and wing. It is felt that this geometry is representative of the advanced technology engine suitable for a second generation supersonic transport.

WEIGHT

The engine weight is based on the GE21/J2B1, scaled to 287 kg/s (633 lbs/sec) airflow. This engine incorporates the same type of nozzle/reverser/suppressor as the J6H2, which was used as the geometry base, and many of the engine design features. However, the J2B1 is significantly lighter than the J6G, due in part to the bypass feature and in part to assumed technology advancements. The preliminary weight estimate for the J2B1 incorporates an estimated allowance for installation factors.

The intake weight is based on the Contractor's N-5 design, with the weight reduced to reflect the deletion of the throat doors. These doors had been incorporated for flow matching in transonic flight with the GE4 J5P and J6G engines. It is assumed that a new technology engine incorporating an integrated inlet engine control system can be flow-matched without the doors, while maintaining the assumed J6G performance.

The total installed engine pod weight is shown in figure 1-5 along with previous installations for reference. The GE4 J5P (1969) engine was used in the original definition of the 969-336C airplane. Subsequent to that definition, the engine, intake and installation weights were revised upward to the levels shown for the J5P (1970). The AST Task III (Contract NAS1-11938) airplane update was based on the GE4 J6H2 large dry turbojet. The weight increase from the J5P (1970) to the J6H2 indicates the weight penalty associated with that approach to meeting the FAR-36 sideline noise requirement. Figure 1-5 shows the ATAT-1 engine installation weights approximately 3400 kg (7500 lb) less than the J6H2 pod; and about 410 kg (900 lb) more than the original J5P (1969) pod.

PERFORMANCE

The performance of the ATAT-1 is based on the GE4 J6G engine, for which computer tapes are available. The GE4 J6G is the most advanced afterburning turbojet defined in the course of the U.S. National SST Program. In overall performance, it compares quite favorably with the advanced technology cycles provided to AST Contractors by the engine manufacturers during 1973.

The installed performance was generated using a Boeing computer program called "Instep", which accounts for all intake drag and recovery losses, ECS and secondary air system losses, bleed and power extraction losses associated with the pod.

In order to be consistent with the assumed weight, the maximum afterburner temperature had to be limited to a value consistent with the annular nozzle of the J2B1. This temperature limit of 1644 K (2960° R) was rigidly adhered to in defining takeoff performance and noise. For supersonic climb, a power setting was used consistent with this limit.

Figures 1-6 and 1-7 show the installed supersonic cruise, subsonic cruise, and climb performance of the engine. Also shown is the comparable performance of the GE21 J2B1 and the P&WA-5B afterburning turbojets at the same design airflow size. The data for the latter two engines are from the preliminary cycle analyses conducted during the

AST Engine Cycle Study Task I (Contract NAS1-11938). Figure 1-6 shows the "advanced technology" cycles have poorer supersonic cruise performance than the ATAT-1 (J6G). It is assumed that cycle matching refinements would bring the advanced cycles to a performance level equivalent to the J6G. Figure 1-6 also shows the advanced cycles are slightly better than the ATAT-1 at a typical subsonic cruise condition, thus partially offsetting the supersonic deficiencies. Figure 1-7 shows that the advanced cycle data generally brackets the ATAT-1 at climb conditions, when each engine is operated at a 1644 K (2960° R) augmentor temperature. It should be noted that, the P&WA-5B engine data does not include boattail drag in this instance. An accurate comparison would show less spread between engines, particularly at the lower Mach numbers.

On the basis of these comparisons, it is felt that the overall ATAT-1 performance, based on the J6G, is representative of an advanced technology supersonic engine.

ENGINE SIZE

To minimize pod weight, the engine is operated at the maximum afterburning temperature of 1644 K (2960° R) at takeoff. At this power setting, an engine size of 287 kg-s (633 lb sec) is required to meet the takeoff balanced field length criterion of 3780 meters (12,400 ft) on a standard +15 K day at sea level. Coincidentally, this is the design size of the J6G and J5P engines, and is a good size match for the airplane.

NOISE AND SUPPRESSION

The ATAT-1 noise levels are based on the J6G engine, for which computer tapes and data were available. The spoke suppressor noise reductions, adjusted for temperature effects, are based on the J6H2 suppressor characteristics. This suppressor was selected due to timing and weight constraints and may not be optimum for an augmented engine. At liftoff speed and takeoff power, the ratio of suppression to thrust loss is approximately "1 for 1"; i.e., the 4 EPNdb suppression results in 4% gross thrust loss.

Thrust and cooling losses associated with the suppressor were assumed to be adequately accounted for in the J6G low speed performance data. Whereas a typical thrust coefficient for the annular nozzle with spoke suppressor deployed would be 0.93 at the takeoff power setting used on ATAT-1, the comparable value used in the J6G data was about 0.88. The 5% difference was assumed to be used up in providing suppressor cooling air that would be required for maximum A B operation. Jet suppressor research has shown promising schemes for operating air cooled spokes in a severe afterburning environment.

At cut-back power conditions over the community and on landing approach, the suppressor is stowed, the inlet is choked, and the aft arc turbo-machinery noise is the principle noise source. The J6G is again the basis for noise estimates at these conditions, including the turbo-machinery noise suppression treatment effect assumed for that engine.

Choked-inlet recovery used at cut-back power, and its suppression effectiveness is taken from National SST Program results. For this study, it is assumed that no operational

restrictions would be applied, i.e., choking could be used at the required power setting, without regard to possible distortion limitations being exceeded.

ENGINE INTEGRATION

DISTANCE AND ALIGNMENT OF ENGINE BELOW WING SURFACE

Intake Diverter Height

To minimize drag, intake diverter heights should be as small as possible while preventing ingestion of the wing boundary layer into the intake. Unpublished Langley wind tunnel data show that boundary layer thicknesses, determined by test, are somewhat thicker than simple theory predicts. The thickness increase varies from 20% thicker for inboard locations to 40% for middle and outboard locations. Thus, the diverter height necessary to assure no boundary layer ingestion at cruise conditions may be based on the theoretical boundary layer thickness for the specific intake location multiplied by the 1.2 or 1.4 factors this NASA data. Theoretical calculations show that, at Reynolds numbers typical of end-of-cruise conditions, the theoretical boundary layer thickness is .012 times the boundary layer development distance. Boundary layer development distance at supersonic cruise is assumed to be the distance from the intake lip to the wing leading edge along a captured streamline flow path.

On the basis of this analysis the diverter heights for the 969-512 are: inboard intake .447 m (17.6 in) and outboard intake .198 m (7.8 in); and for the 969-513 are: inboard intake .544 m (21 in) and, outboard intake .333 m (13.1 in). The large diverter heights on the inboard intake require either extremely deep gullies in the upper surface of the wing to allow proper aft nacelle fairing and reverser exhaust over the wing, a change in wing undersurface reflex, or the use of pod struts.

Also, the unpublished Langley data shows that for intakes located close inboard very large increases in boundary layer thickness will result at moderate angles of sideslip. The boundary layer thickness at the inboard intake of configuration 969-513 will increase to about .889 m (35 in) at .017 radian (1 degree) to .035 radian (2 degrees) sideslip angle during Mach 2.7 flight. This can result in compressor stall and intake unstart. A possible solution to avoid intake unstart and compressor stall during this transient condition is to develop very high response intake/engine controls with an integrated engine-intake control system to increase the compressor stall margin. The outboard intake of 969-513, and both intakes of 969-512 will be free from excessive boundary layer thicknesses during sideslip conditions.

Intake and Engine Alignment

A linearized supersonic flow field analysis has been used on earlier SST configurations to determine the local flow direction to align the intake at supersonic cruise. In the absence of such analysis for the current configurations, the sidewash alignment angles in the plan view are estimated from these previous analyses. Propulsion pod alignment angles in the plan view are dependent upon the distance from the airplane centerline. Relative to the airplane centerline the pod alignment angles for the 969-512 are:

inboard pod .023 radian (1.3 degrees) and outboard pod .040 radian (2.3 degrees); and for the 969-513 are: inboard pod .007 radian (0.4 degrees), and outboard pod .028 radian (1.6 degrees). The intake alignment angle in the side view is assumed to be equal to the wing surface angle at a point just ahead of the intake lip. This point is located by projecting a ray forward from the center of the intake at the lip station, at an angle equal to the Mach angle until it intersects the wing surface. This angle is .384 radian (22 degrees) for Mach 2.7.

SEPARATION AND STAGGER OF ADJACENT ENGINE NACELLES

The intakes must be separated sufficiently to preclude mutual upstarts. Testing done during the National SST Program shows this is possible if the ratio of spacing between intake centerlines at the lip station is = 2.84 times the intake diameter and the intakes are within one-fourth of an intake diameter of being coplanar.

JET FLOW IMPINGEMENT

The most serious jet flow impingement problem is associated with the inboard engine and the horizontal tail at Mach 2.7 with high angles of attack and high engine exhaust temperatures. Impingement studies of Models 969-512 and 969-513 have been conducted using data from reference 1-3. These data neglect the effect of local airplane flow fields even though these are known to have a significant effect on jet flow field, and must be considered for more in-depth configuration development. Based on reference 1-3 data, during high angle-of-attack maneuvers with the tail and elevators fully rotated, the jet flow from the inboard engine of the Model 969-512 passes outboard of the horizontal tail, and the jet flow from the inboard engine of the Model 969-513 passes under the horizontal tail.

MINIMUM DRAG LOCATION

A significant amount of effort has been devoted to the establishment of guidelines for engine airframe integration for large supersonic airplanes. Figure 1-8 illustrates the results of such a study. Work of this nature leads to the conclusion that minimum drag installation is achieved when the pod inlets are located behind the line of maximum wing thickness and the maximum pod area is close to the wing trailing edge. Lateral pod location is determined by spacing required to avoid mutual pod inlet interference as discussed previously and does not have a significant effect on cruise drag. These principles were used in locating the pods for both the Model 969-512 and Model 969-513 airplanes.

CONFIGURATION SELECTION

The two configurations are evaluated as discussed in the following paragraphs. In tables 1-1 and 1-2 the primary characteristics of the two configurations are defined. When the "base" is used in the 969-513 column the corresponding value in the 969-512 column shows the difference between the two configurations. The comment column gives a brief explanation of the impact of the differences between the configurations. The final

selection is made on the basis of best range performance as discussed in the configuration selection later on in section 1.

LANDING GEAR INDUCED SLUSH, WATER AND FOREIGN OBJECT INGESTION

Because the engine intakes are located behind the main landing gear the intakes will be vulnerable to foreign material ingestion and a rather complex fender and deflector system on the landing gear is required. Also the landing gear door arrangement and deployment sequence will need special design attention to assure that the intake will be protected from foreign material blown off the landing gear and doors during landing gear retraction and extension. The intake ingestion characteristics of the Model 969-512 are inherently superior to the Model 969-513 because the inboard engine is outboard of the landing gear on the 969-512.

SONIC NOISE ON ADJACENT STRUCTURES AND COMPONENTS

Studies conducted during the National SST Program showed that the near field sonic noise problem was almost entirely during the takeoff portion of the mission. Near field noise data were obtained from full scale GE4 engine tests at General Electric. These data were applied to surfaces on the 969-512 and 969-513 and are shown in figure 1-9.

LOW SPEED CHARACTERISTICS

Both the 969-512 and 969-513 closely resemble the wind tunnel models tested by Langley. Differences exist in wing trailing edge geometry, engine location and forebody length. These differences are believed to have minor impact on longitudinal stability and control. Therefore no attempt is made to account for these differences when balancing the two configurations.

Low speed lift and drag were estimated using parameters derived from the Langley test mentioned above plus data from NASA Ames. The Langley test data was also used to provide a flaps up, leading edge down, base configuration. Increments due to flap deflections for the study configurations were then added to this base.

A comparison of pertinent low speed characteristics are tabulated below:

<u>Configuration</u>	<u>δF, APP</u>	<u>α LOF</u>	<u>L/D APP</u>
969-512	.262 rad(15°)	.154 rad(8.8°)	4.4
969-513	.262 rad(15°)	.161 rad(9.2°)	4.9

Low speed flaps down L D was better for the 969-513 than for the 969-512 resulting in somewhat lower noise levels during takeoff and landing operations. The 969-513 on the other hand requires a higher liftoff angle of attack resulting in a longer landing gear.

Low speed lateral control is superior on the 969-513 although both configurations meet the low speed roll control criteria.

The 969-513 engines are located closer to the airplane centerline than is the case on the 969-512. Following critical engine failures during takeoff the thrust moments generated on the 969-513 will be less than those on the 969-512. Therefore, a smaller body-mounted vertical is used on the 969-513 resulting in less weight and drag.

COMPARATIVE HIGH SPEED CHARACTERISTICS

Aerodynamic characteristics in cruise have been analyzed for the 969-512 and 969-513 airplanes at Mach 2.7 and an altitude of 18,288 m (60,000 ft). The drag assessment considered zero lift wave drag, friction drag and drag due to lift including nacelle influence. The fuselage of both models has been optimized to achieve minimum wing-body wave drag. The lift drag characteristics are shown in figure 1-10.

Cruise pitching moment data were also estimated and are shown in figure 1-10. The theoretical moment curves were corrected by wind tunnel increments for similar planforms to obtain the data shown. The trim drag increments shown in the upper right hand corner of the figure are based on these moment curves.

STABILITY AND CONTROL

This section discusses stability and control criteria used in the study. Empennage sizing and lateral control sizing are discussed, as are the automatic unstart system, the alpha limiter system, and aeroelastic effects.

Longitudinal Stability and Control Criteria

The longitudinal stability and control criteria used in this study are outlined below. These criteria are based on the criteria defined in reference 1-4.

The horizontal tail shall be sized to provide adequate control power for takeoff rotation, landing flare and stall recovery. Any deficiencies in stability throughout the flight envelope shall be compensated for through the use of stability augmentation.

The aft center of gravity limit must be selected such that available control power is sufficient to generate a 0.1 rad/sec^2 nose down pitch acceleration for stall recovery at $V_{\text{MIN DEM}}$ (ref. 1-5).

The forward cg limit is set by the most demanding of the following criteria:

- At $V_{\text{MIN DEM}}$ it shall be possible to trim the airplane for straight and level flight in a landing configuration. The lift coefficient ($C_{L\text{MAX DEM}}$) corresponding to $V_{\text{MIN DEM}}$ is 50% larger than $C_{L\text{APP}}$ resulting in a 0.5 g maneuver load margin at V_{APP} .

- At takeoff it shall be possible to rotate at a rotation speed ($V_R = 180$ kts) as discussed in reference 1-6.
- At landing flare the airplane shall meet the pitch acceleration criteria shown in figure 1-11.

Lateral-Directional Stability and Control Criteria

The lateral directional stability and control criteria used in this study are outlined below. They are also based on criteria in reference 1-4.

Directional controls are sized for engine-out control. Lateral and directional forces and moments from engine failures must be balanced—steady state—with a maximum of .175 radian (10°) of rudder deflection and .052 radian (3°) of sideslip.

At takeoff the yawing moment caused by an outboard engine failure shall be balanced by rudder alone at the engine-out balance speed (V_{BAL}). The engine-out balance speed is defined as

$$V_{BAL} = V_1 + 15 \text{ knots.}$$

Lateral controls are sized for roll response and engine-out control. Roll response performance is expressed in terms of a required bank angle change in a given time (ϕ_t). In the takeoff, approach, and climb configurations the bank angle change shall be .524 rad (30°) in 2.5 seconds ($\phi_{2.5} = .524$ rad). In the climb, cruise, and descent configurations, at all speeds up to V_{MO}/M_{MO} it shall be possible to achieve a bank angle of 1.047 rad (60°) in 7 seconds ($\phi_7 = 1.047$ rad).

During the transient following an engine failure maximum available roll control may be used. To trim the airplane after the transient has subsided a maximum of 66% of the available roll control may be used.

Longitudinal Stability and Control Evaluation

The planforms of the 969-512 and 969-513 are similar. Differences exist in wing trailing edge geometry, engine location and forebody length (see fig. 1-12). These differences will only have minor impact on basic wing-body stability and are ignored in this preliminary configuration stage.

Unpublished Langley wind tunnel data were used to establish basic wing-body low speed stability. The two configurations have horizontal tail geometries identical to that of the 1971 U.S. SST airplane. This horizontal tail configuration was chosen rather than those tested by NASA Langley because test data show that it has a steeper lift curve slope and a greater CL_{MAX} (ref. 1-6). Horizontal tail control power estimates were based on data in reference 1-6. These include aeroelastic effects.

The aft cg limit was selected such that available control power is sufficient to generate a 0.1 rad/sec^2 nose down pitch acceleration at $V_{MIN DEM}$. A margin ($\Delta C_M = 0.01$) was

subtracted from the estimated available pitching moment coefficient to account for data uncertainties. An alpha limiter system (see paragraph later on in section 1) is used to prevent flight at angles of attack greater than that corresponding to $C_{LMAX DEM}$.

The forward cg limit was selected such that the airplane meets the trim requirements at $V_{MIN DEM}$ and satisfies the nose wheel lift off criteria at the rotation speed ($V_R = 180$ knots).

In length units, cg travel, was assumed the same as for the 969-336C airplane (ref 1-1). Minimum tail sizes required to meet these cg ranges are shown in figure 1-13 with forward and aft cg limits plotted as functions of the tail volume coefficient. The cg range requirements between the aft cg limit and the forward limit for takeoff rotation is sizing, resulting in a $V_{HMIN} = .0435$.

Lateral-Directional Stability and Control Evaluation

The area of each of the wing mounted vertical surfaces is chosen so that the "wing-body-wing verticals" directional stability at high speed ($M = 2.7$, $q = 36580.3 \text{ N/m}^2$ (764 psf) is neutral ($C_{n\beta} = 0$) at cruise α , the same as for the Langley SCAT-15-F-9898 wind tunnel model. This is achieved by keeping the ratio between the vertical surface area and the gross wing area the same on the "Arrow Wing" as on the Langley wind tunnel model.

The centerline vertical surface has the same geometry as that of the 1971 U.S. SST and is sized for engine-out control. At high speed ($M = 2.7$, 1 g flight), the yawing moments from an outboard engine seizure and inboard engine inlet unstart are balanced steady state with 0.175 rad (10°) of rudder deflection and 0.052 rad (3°) of sideslip. The aerodynamic moments due to this failure mode are assumed to be the same as for the 1971 U.S. SST (see ref. 1-7), but scaled to new engine locations. In addition, asymmetric thrust has to be balanced. The change in thrust level due to this failure is also assumed to be the same as for the 1971 U.S. SST (see ref. 1-7). Figure 1-14 shows tail volume coefficient sensitivity to shifts in engine spanwise location. The sizing of the Arrow Wing Model 969-512 is shown as an example.

At low speed (takeoff) the yawing moment from an outboard engine failure is balanced by rudder alone. The V_1 speed chosen is the same as for the 2707-300 at $340,200$ kg ($750,000$ lb). ($V_1 = 157$ knots). The centerline vertical surface is sized to statically balance the engine-out moment at V_{BAL} ($V_{BAL} = V_1 + 15$ knots). Engine differential thrust (net thrust-inoperative engine drag) is assumed to be $249,088$ N ($56,000$ lb) (see ref. 1-7). The centerline vertical surface yawing moment coefficients due to sideslip and rudder deflection are based on 1971 U.S. SST data (ref 1-6). These data include aeroelastic effects. Table 1-3 summarizes the centerline vertical surface sizes for the two configurations.

The high speed lateral control evaluation was conducted only on the 969-512 configuration due to time limitations. Although the engines on the 969-513 configuration are further inboard and the outboard wing has two spoiler slot deflectors instead of one spoiler and one spoiler slot deflector it has been assumed that the

conclusions drawn from the 969-512 high speed lateral control evaluation would also apply in the case of the 969-513.

The high speed lateral controls for the 969-512 configuration consist of an inboard flaperon, an inboard and outboard plain spoiler, and outboard inverted spoiler slot deflector. The estimated rolling moment contributions of the individual control surfaces are presented in table 1-4. These estimates are based on 1971 U.S. SST wind tunnel data where possible. Configuration differences are accounted for; such as differences in surface area moments, reference areas and dimensions, and hinge line sweep. Lacking wind tunnel data, control power was estimated using methods described in reference 1-8. The control surface moment coefficients are corrected to include aeroelastic effects.

Engine-out disturbance characteristics and aerodynamic interference effects are based on 1971 U.S. SST data (ref. 1-7) with corrections applied for engine location. Thrust effects were computed using a maximum thrust of 116,538 N (26 200 pounds). Engine failed thrust is also assumed to be the same as for the 1971 U.S. SST. A summary of the engine-out moments components used in this analysis is presented in table 1-5.

Two basic types of engine failures were considered. They were an outboard engine seizure in combination with an inlet unstart and total flameout of the adjacent inboard engine, or an inlet unstart and total engine flameout of both engines on the same side. The engine seizure case was only considered at 1.0 g flight due to the low probability of occurrence while the double unstart, with a relatively high probability of occurrence, was considered up to 2.0 g's.

Satisfactory control exists for the 1.0 g engine seizure where the aircraft exhibits low dihedral effects. The greatest roll control requirement occurs at the instant of engine failure, before any sideslip angle has developed. As sideslip increases to the peak estimated value of 0.075 rad (4.3°) the roll due to dihedral becomes predominant and opposes the roll due to the engine failure. The required roll control at peak sideslip is approximately 13 percent of available control. At a steady state sideslip, $\beta = 0.052$ rad (3°), the reduced dihedral effect results in an increase in required control up to 22 percent of that available. This is summarized in figure 1-15.

At 2.0 g's the airplane exhibits large dihedral effects and a decrease in directional stability. This results in an unacceptable engine failure case where insufficient control exists to counter the steady state condition, much less control the peak overshoot condition. The large control requirements are due predominantly to the high dihedral effect exhibited by the configuration at the angle of attack required for 2.0 g flight.

Three approaches to control of the 2.0 g double unstart case were considered:

- Increase directional stability
- Increase roll control power
- Control engine failure with an automatic restart system

To increase directional stability, an increase in vertical tail size is required with a corresponding increase in drag and weight. This is an unacceptable approach. An increase in lateral control would require a significant increase in lateral control surface area. Insufficient room exists on the wing to increase the area of the lateral control surfaces. Also, large lateral control power can be detrimental at high Mach numbers. Maneuvering at load factors up to 2.0 g's at high roll rates with low directional stability can result in excessive sideslip even with no engine failure. This case is described in reference 1-9. In this study an automatic restart system, is assumed (see following paragraph). The effect of this system on the 2.0 g double unstart engine failure is significant. With the automatic restart system the airplane meets the engine-out control criteria (see fig. 1-15).

The roll response criteria for normal operation during climb, cruise, and descent requires that a 1.0 ± 7 rad (60°) bank angle change be achieved in 7.0 seconds. This was found to be most critical at $M = 1.5$. The same flight condition is assumed critical for the 969-512. The required roll control power to meet this criteria with the 969-512 configuration is 75 percent of that available at this flight condition (see fig. 1-15).

Automatic Unstart System

As discussed in the previous section, an automatic unstart system to reduce lateral-directional forces and moments is required. Alternate solutions such as increased lateral control power and larger vertical surfaces are impractical. The automatic unstart system will sense the presence of an outboard engine inlet unstart in combination with a lateral acceleration. When this occurs it will initiate a restart cycle on the opposite engine inlet, thereby reducing the yawing and rolling moments caused by the initial engine failure. The cycle will be terminated when the inlet is stabilized in the unstarted condition. Pilot action will then be required to restart the inlets.

The longitudinal deceleration from the activation of the unstart cycle is small. To illustrate this, time responses were calculated for a two-engine flameout during a 2 g maneuver. It was assumed that the pilot would bring the airplane back to a 1 g flight condition within 10 seconds of the engine flameout. Figure 1-16 shows load factor, longitudinal deceleration, true airspeed, and Mach number versus time for the flameout without automatic unstart. At the time of the flameout the airplane is already decelerating at a rate of 1.219 m/sec^2 (4 ft/sec^2). The peak deceleration, 1.98 m/sec^2 (6.5 ft/sec^2), occurs 0.5 seconds after the flameout. The time history when an automatic unstart system is used is shown in figure 1-17. The increase in longitudinal deceleration is $.305 \text{ m/sec}^2$ (1 ft/sec^2) or 0.03 g's. This increase in longitudinal deceleration is small and should be of little concern when implementing an automatic unstart system.

Alpha Limiter System

Available flaps-down wind tunnel data for arrow wing configurations exhibit an unstable pitch break, although the magnitude is dependent on the leading edge flap or slat configuration, leading edge radius, basic wing planform, and trailing edge flap or control configuration. The existence of this characteristic would, in general, require a forward shift of the cg range if an alpha limiter system were used.

The data available for the clean (flaps-up) configuration are not sufficient to define adequately the aerodynamic pitch-up characteristics.

In addition to these longitudinal considerations, there are lateral-directional considerations for angle of attack limiting. Because of its impact on engine failure control in supersonic flight, an alpha limiter makes possible the development of a configuration with less directional control power. In general, supersonic airplanes exhibit deterioration of directional stability with increasing angle of attack. Also, the probability of an inlet unstart increases with angle of attack. Limiting the cruise angle of attack to that corresponding to 2 g therefore permits control of engine failures with smaller directional control surfaces, which in turn leads to better cruise performance.

A conceptual block diagram of the selected alpha limiter system is shown in figure 1-18. The system will determine if, and at what rate, alpha limit is being approached and provide the nose down command required for safe recovery. The alpha limiter system will not affect airplane characteristics until alpha limit is being approached. At this time it will modify the apparent airplane pitching moment by commanding nose down elevator if the pilot has not already initiated recovery. A more detailed discussion of the alpha limiter system is published in reference 1-5.

Aeroelastic Effects

Aeroelastic effects on wing-body stability at low speed have not been included because they are small at that condition.

Control power estimates for longitudinal, lateral, and directional control surfaces are based on 1971 U.S. SST data. These data include the effects of structural flexibility.

COMPARATIVE WEIGHT AND BALANCE ASSESSMENT

Final definition of the Model 969-512 "outboard engine" airplane and the Model 969-513 "inboard engine" airplane was partially dependent on airplane balance and the resulting weight effects. Both airplanes have been configured with identical 55.74 m² (600 ft²) horizontal tails and horizontal tail arms of 26.97 m (1062 in.), resulting in the same horizontal tail V and airplane center of gravity limits. The further aft location of the outboard engines on the 969-512 requires greater forward body length to maintain the same balance as the 969-513. The longer forward body and shorter engine to vertical tail distance requires a larger vertical tail on the 969-512. The weight increases on the 969-512 for the longer forward body and larger vertical tail are more than offset by the weight decrease due to the increased wing box depth and shorter landing gear as compared with the 969-513 airplane. The weight increments for these design differences between the two airplanes are shown in table 1-6.

PERFORMANCE SENSITIVITIES

The sensitivity to OEW and cruise L/D was evaluated on the 969-512B configuration, figure 1-19. A 1% change in OEW corresponds to 2% of the range, while a 1% change in L/D corresponds to 1.1% of the range.

COMPARATIVE PERFORMANCE

The 969-512 configuration with pods located outboard has 306 km (165 nmi) more range than the -513 configuration with pods located inboard, table 1-7. Low speed performance and noise are essentially the same for both airplanes with the exception of the approach noise. The 969-513 configuration had 3 EPNdB lower approach noise due to a higher L/D. The L/D improvement is due mainly to less trim drag on the 969-513 at the expense of a longer gear compared to the 969-512.

CONFIGURATION SELECTION

The 969-512 was selected as the best configuration for the structural analysis based on the lower operating empty weight and better range. Several trade studies were conducted to optimize this configuration.

CONFIGURATION REFINEMENT AND DEFINITION

A new baseline configuration evolved from the 969-512. This new configuration, the 969-512B, shown in figure 1-20 differs from the original 969-512 by a larger flap deflection, new cg limits, shorter landing gear, and a forward body shift as discussed later on in section 1.

HORIZONTAL TAIL SIZE STUDY

A study was undertaken to determine the airplane weight sensitivity due to variations in horizontal stabilizer size. Table 1-8 shows the parameter differences and resulting weight changes for a 55.74 and 44.50 m² (600 and 479 ft²) horizontal stabilizer. The smaller stabilizer satisfies the control power requirements, but the larger stabilizer results in a lighter OEW and permits a more aft center of gravity range, thereby improving the airplane loadability. Consequently, a 55.74 m² (600 ft²) horizontal stabilizer was retained for the 969-512B configuration.

LONGITUDINAL STABILITY AND CONTROL

The horizontal tail sizing chart for the 969-512B is shown in figure 1-21. This chart is based on updated low speed stability and control characteristics. The minimum tail area that would meet cg range requirements is 50.44 m² (543 ft²). However, a 55.74 m² (600 ft²) tail was chosen as stated above.

A check of pitch acceleration capability in ground effect shows (see fig. 1-22) that at the design CL_{App} the 969-512B has marginal pitch acceleration capability at maximum landing weight 217,728 kg (480,000 lb). At lower landing weights it is unacceptable. This problem is solved by operating the airplane at a fixed approach speed corresponding to the maximum landing weight, regardless of actual weight, thereby maintaining the required dynamic pressure for adequate horizontal tail control power.

Figure 1-23 shows the variation of aft cg limit with $C_{N_{MAX DEM}}$ for the 969-512B in a clean (flaps up) configuration and with leading edge devices drooped. This figure indicates that the airplane does not have adequate stall recovery capability when flown at the aft cg limit at the selected $C_{N_{MAX DEM}} = .56$ in a clean configuration. However, it does with leading edge devices drooped. Therefore, flaps and leading edge devices will be deflected as required during subsonic flight to assure adequate stall recovery.

AUGMENTATION SYSTEM DEFINITIONS

To meet minimum safe operation criteria (ref. 1-4) throughout the flight envelope, a flight critical augmentation system (HSAS) will be used. A functional diagram of the longitudinal augmentation system is shown in figure 1-24. Briefly it consists of a pitch rate feedback signal that is shaped in the HSAS filter to provide increased aerodynamic stiffness and damping. A second order structural mode filter is used to prevent coupling between airplane dynamics and structural modes.

Figure 1-25 shows a functional diagram of the lateral-directional augmentation system. In this system yaw rate and lateral acceleration are fed back to rudder for dutch roll damping and sideslip control following engine failure, respectively. Yaw rate to aileron feedback is used to reduce the divergence rate of the spiral mode and roll rate to aileron is used to improve the roll time constant. Structural mode filters are used in all feedback loops to prevent coupling between airplane dynamics and the structural modes.

WING TWIST STUDY

A study was made to determine the effect of wing twist on the 969-512 cruise drag. Preliminary evaluation of the cruise aerodynamic characteristics of the 969-512 indicates that large values of trim drag may be incurred when flying at cg's near $.505 \bar{c}$. The following is the result of an investigation to determine if this penalty can be reduced by adding negative twist, i.e., wash-out, to the wing outboard of the inboard nacelle. The results are shown in figure 1-26.

The variation of drag with trimming moment produced by the horizontal tail is shown in the upper left hand portion of the figure for the 969-512 airplane with leading edge droop. The point noted on this curve for the cg at $.505 \bar{c}$ is the reference for the study and all drag increments were taken relative to it. The upper right hand portion indicates the variation of C_{M_0} with wing twist. The wing leading edge was twisted down linearly from $34.7\% b/2$ to the wing tip and the resulting moment and drag increments were calculated. The lower left hand inset shows the drag increment due to the additional wing twist.

The influence of wing twist on trim drag and total drag is illustrated in the remaining two sets of curves for several cg locations. These data indicate that outboard wing twist from $-.035$ to $-.070$ rad (-2° to -4°) would be highly beneficial. However, consideration of the effect on takeoff aerodynamics would dictate less twist. An incremental twist angle of about $-.026$ rad (-1.5°) is a good compromise which results in a 5.5 to 8 drag count reduction, depending on the cg location, and a lift-off angle increase of about $.003$ rad (0.2°).

ADDITIONAL CONFIGURATION REVISIONS

The remaining configuration revisions between the Model 969- 512 and the 969-512B are a result of the following design refinements. A .087 rad (5°) increase in flap setting to .349 rad (20°) gives an improvement in low speed lift, a reduction in liftoff angle of .012 rad (0.7°) and can be trimmed with the 55.74 m^2 (600 ft^2) horizontal stabilizer. In addition, it is possible to move the center of gravity range aft by 1.5% of the reference chord and shorten the length of the main landing gear oleo. The rebalance of the Model 969-512 with these new limits requires the body shifted .445 m (17.5 in.) (one body frame spacing) aft on the wing. The resulting configuration is designated Model 969-512B as shown in figure 1-20.

FUEL TANK ARRANGEMENT

The temperature of the fuel in the tanks must be kept low enough so that cavitation does not occur at the boost pumps. Current estimates show that at the high altitudes occurring during supersonic cruise, the average temperature of the fuel in the tank must remain below (344 K) 160° F for unpressurized tanks.

The rise in temperature of the fuel for the basic fuel tank arrangement used on the 1968 versions of the Arrow Wing (969- 336C, ref. 1-1) was analyzed during the AST Task III system study (ref. 1-5) for typical flight missions for various fuel tank wall thermal conductances. These studies show that the rear main tanks, i.e., tanks 2 and 3 in figure 1-27, are especially vulnerable to aerodynamic heating. The effect of fuel tank wall conductance on the fuel temperature at the end of cruise is shown in figure 1-28. This shows that the rear main fuel tanks together with the auxiliary tanks which feed them during supersonic cruise should have tank wall thermal conductances less than $14.2 \text{ W/m}^2 \text{ K}$ ($2.5 \text{ Btu/hr. ft}^2, ^\circ \text{F}$). The forward main tanks are less vulnerable to heating but should have tank wall thermal conductances less than $38.4 \text{ W/m}^2 \text{ K}$ ($6.77 \text{ Btu/hr. ft}^2, ^\circ \text{F}$). This extremely low conductance requirement led to the investigation of other fuel tank arrangements to reduce the amount of insulation required. The AST Task III study (Contract No. NAS1-11938) indicates that it is desirable to locate the fuel tanks in the deeper sections of the wing and body.

In addition to the fuel temperature problem, the fuel tank boundaries have been moved to position the center of gravity within the airplane center of gravity limits. To alleviate both of the above concerns, the fuel tanks are moved inboard and an auxiliary tank is added to the aft fuselage. The tank layout evaluations consider the following requirements:

- The cg of the airplane in the zero usable fuel condition shall be within the operational landing cg limits for all payload conditions. Fuel ballast may be used to place the airplane cg within these limits for partial payload conditions.
- Fuel loading shall place the airplane cg within takeoff limits and fuel management shall maintain the airplane cg within flight limits.

- Airplane cg shall be controlled by the transfer of fuel from the fore and aft auxiliary fuel tanks.
- There shall be one main tank for each engine and the engines shall be continuously fed from these tanks.
- The main tanks shall be sized with as near identical capacity as practical.
- Adequate fuel quantity in each main tank shall be maintained during fuel management to prevent inadvertent tank depletion.
- Reserve fuel shall be contained in the main tanks during landing.
- Fuel from the auxiliary tanks shall be transferred into the mains. Once an auxiliary tank is switched on, pumping shall continue until the tank is empty.
- Fuel loading and usage sequence for varying payloads shall be similar, to simplify fuel management.

The total tank capacities are more than adequate to permit a mission starting at a maximum taxi weight of 340,194 kg (750,000 lb) for any desired payload. The fuel tank arrangement selected for the 969-512B configuration is shown in figure 1-29. The corresponding fuel management scheme is shown for the two extreme payload conditions in figure 1-30.

MODEL 969-512B WEIGHT, BALANCE AND PERFORMANCE

The Model 969-512B is 721 kg (1590 lb) lighter than the original 969-512, as shown on table 1-9.

A group weight and balance statement for the Model 969-512B airplane is tabulated in table 1-10.

The Model 969-512B has a range improvement of 53.7 km (29 nmi) over the original 969-512, as shown in table 1-7.

ALTERNATE ENGINE LOCATIONS

Three alternate locations are provided to assess structural sensitivity to small changes in engine location. At the onset of Task I it was planned to conduct flutter analyses during Task III with engines at various locations along the trailing edge to obtain flutter speed sensitivity to engine position. Subsequently, this Task III effort was deleted. The rationale for establishing the various location however was addressed in Task I and is reported below.

Flutter speed is generally sensitive to the size and weight of the outboard engine and its location relative to the wing structural box. In order to investigate the influence on flutter, a configuration is drawn with the outboard engine moved forward one-half engine inlet diameter, relative to the position shown on Model 969-512B (fig. 1-31).

One-fourth engine intake diameter is the maximum stagger permissible to avoid mutual engine inlet unstart. For performance reasons the inboard engine is kept at its position.

In order to investigate the spanwise influence of the engine's location with respect to flutter, a configuration was drawn with both engines moved inboard 1.52 m (60 inches) (fig. 1-32). A secondary benefit from moving the engines inboard is that the size of the outboard spoiler/slot deflector can be increased and inboard spoiler removed. This reduces the change of buffet that could result from having to use an inboard spoiler.

With the outboard engine moved inboard 1.85 m (73 inches) and the inboard engine moved outboard .64 m (25 inches) forming a dual engine pod a third alternate power plant location is created. The longer, two-dimensional inlets are heavier and have a cg located farther forward than the sum of the individual, single pods. It is questionable whether the total airplane performance can be maintained with a dual pod so designed. This configuration is shown in figure 1-33.

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TABLE 1-1.-CONFIGURATION DEFINITION

Subject	969-512		969-513		Comments
	Eng unit	SI unit	Eng unit	SI unit	
Wing area (reference)	9,848 ft ²	914.91 m ²	9,848 ft ²	914.91 m ²	Caused by engine placement TE forward on -513 2.29 m (90 in.) more empty body in -512
Wing area (gross)	11,244 ft ²	1,044.60 m ²	10,220 ft ²	949.47 m ²	
Body length	303 ft, 4 in.	92.46 m	295 ft, 10 in.	90.17 m	Engines farther outbd on -512 Inbd engine lower -513
Horizontal tail area	600 ft ²	55.74 m ²	600 ft ²	55.74 m ²	
Vertical tail area (wing mtd)	287 ft ²	26.66 m ²	287 ft ²	26.66 m ²	Engine farther forward on -513 because of inbd location Structural box farther aft on -513
Vertical tail area (body mtd)	449 ft ²	41.71 m ²	305 ft ²	28.34 m ²	
Landing gear length (static)	211 in.	5.36 m	249 in.	6.32 m	Farther fwd inbd TE causes longer box on -513
Engine loc (spanwise) inbd/outbd	257 in./438 in.	6.53 m/11.13 m	92 in./303 in.	2.34 m/7.70 m	
Engine loc (chordwise) inbd/outbd (Δ)	98 in./116 in. (aft)	2.49 m/2.95 m (aft)	Base	Base	
Wing box depth (mid spar depth)	46.1 in.	1.17 m	43.5 in.	1.10 m	
Wing box chord (SOB)	385 in.	9.78 m	285 in.	7.24 m	
Wing box length	1,077 in.	27.36 m	1,135 in.	28.83 m	

TABLE 1-2.-CONFIGURATION ASSESSMENT

Subject	969-512	969-513	Comments
Thrust reverser installation	Similar to 2707-300	Installation problem on inbd eng	Proximity to body will limit reverser use on -513
Access for galley servicing	Marginal	Marginal	Same risk on both configurations
Spoiler/slot deflector location	Inadequate control power	Adequate control power	Would require increased area and/or moment arm
Plume effects	None	Takeoff rotation only	Possible lower aft body and horizontal tail exposure on -513
Acoustic environment (structural)	168 dB--horiz tail 164 dB--aft body	169 dB--horiz tail 166 dB--aft body	Heavier structure required on -513
Boundary layer--inlet effects	0.45 m (17.6 in.) boundary layer at inbd inlet	0.53 m (21 in.) boundary layer at inbd inlet	Diverter height greater on the -513 (risk on increase in boundary layer with yaw angle)
Engine burst; cabin vibration	Less cabin exposure to burst--lower vibration level	Base	Longer structure path on -512 should reduce engine--induced vibration levels
Aeroelastic effects on S and C	Less detrimental than -513 due to stiffer wing	Base	-512 will have farther aft limit at cruise--less trim drag
Flutter characteristics	Lower flutter speed than -513	Base	Outbd engs on -512 give lower flutter speeds
Foreign object ingestion	Acceptable	Marginal	inbd eng proximity with landing gear retraction on -513 present ingestion problem
Height of passenger cabin from ground	6.68 m (263 in.) between cabin floor and ground	7.75 m (305 in.) between cabin floor and ground	Difficulty in emergency egress with increased height
Cruise drag	Slightly lower drag	Base	Because of less t/c of inbd chord and smaller diverter area
Operating empty weight	-2,327 kg (-5,130 lb)	Base	See table 1-3

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TABLE 1-3.—CENTERLINE VERTICAL TAIL SIZE

Model	$\bar{V}_v = \frac{S_v l_v}{S_w b_w}$	Critical design condition
969-512	0.028	Engine out at takeoff (V_1)
969-513	0.019	Engine out at takeoff (V_1)

*Tail moment arm measured from the aft cg limit at takeoff (see figure 1-14).

TABLE 1-4.—969-512 ESTIMATED LATERAL CONTROL POWER

$$\left(C_l = \frac{\text{rolling moment}}{qSb} \right)$$

Flight condition	Inboard flaperon	Inboard spoiler	Outboard spoiler	Inverted spoiler-slot deflector	Total control available
Mach 1.5, V_{MO} climb wt roll response (1.047 rad in 7 sec)	0.00066	0.00072	0.00028	0.00035	0.00201
Mach 2.7, V_{MO} , $n_z = 1.0$ engine failure (outbd seizure, inbd unstart)	0.00041	0.00035	0.00024	0.00037	0.00137
Mach 2.7, V_{MO} , $n_z = 2.0$ engine failure (double unstart)	0.00045	0.00053	0.00025	0.00037	0.00160

TABLE 1-5. — MAGNITUDE OF ROLLING MOMENTS WITH ENGINE FAILURES, 969-512

$$C_l = \frac{\text{rolling moment}}{qSb}$$

Failure condition	Engine-out aerodynamic			Dihedral effect			Total		
	Initial	*Peak	Steady state	Initial	*Peak	Steady state	Initial	*Peak	Steady state
Mach 2.7, V _{MO} n _z = 1.0 Outboard seizure inboard unstart	0.0013	0.00169	0.00136	0.0/0°	-0.00151/β = 0.075 rad (4.3°)	-0.00105/β = 0.052 rad (3°)	0.0013	-0.0018	0.00031
Mach 2.7, V _{MO} n _z = 2.0 Double unstart (w/o automatic restart)	0.00098		0.00091	0.0/0°		-0.00270/β = 0.063 rad (3.6°)	0.00098		-0.00179 (exceeds roll control power)
Mach 2.7, V _{MO} n _z = 2.0 Double unstart (w/automatic restart)	0.00098	0.00034	0.00034	0.0/0°	-0.00151/β = 0.035 rad (2.02°)	-0.00105/β = 0.024 rad (1.4°)	0.00098	-0.00117	-0.00071

*Peak condition occurs approximately 3 sec after engine failure

TABLE 1-6.—WEIGHT DIFFERENCES, MODEL 969-513 TO MODEL 969-512

	OEW (kg)	OEW (lb)
Model 969-513	164,119	361,820
Wing box		
Decrease elastic axis length 5%	-1,315	-2,900
Increase side of body depth 6% and increase box root chord length. (No allowance for possible differences in flutter and engine dead weight relief)		
Wing trailing edge	-73	-160
Area differences		
Vertical tail and controls	+753	+1,660
Increase area from 28.34 m ² (305 ft ²) to 41.71 m ² (449 ft ²)		
Body and systems	+1,370	+3,020
Increase forward body length 2.29 m (90 in.)		
Entry doors	-231	-510
Decrease width of 4 doors from 1.07 m (42 in.) to 0.81 m (32 in.), and move 4 doors 6.10 m (240 in.) forward		
Nacelle diverter	-77	-170
Decrease depth		
Landing gear	-2,377	-5,240
Decrease length 0.97 m (38 in.), eliminate canted trunnion, and decrease fenders		
Escape slides	-59	-130
Decrease length.		
Aft Body	-318	-700
Estimated reduced sonic fatigue and wing rear spar 2.54 m (100 in.) aft		
Net Weight Change	(-2,327)	(-5,130)
Model 969-512	161,792	356,690

TABLE 1-7.-BOEING/NASA ARROW WING TASK I SST PERFORMANCE SUMMARY

- Takeoff gross weight: 340,194 kg (750,000 lb)
- Payload: 22,183.8 kg (48,906 lb); 234 passengers, tourist
- Cruise mach: 2.7 standard day

	969-512 1% LE rad + LE droop	969-513 1% LE rad + LE droop	*969-512B 1% LE rad + LE droop
Propulsion ● Type ● Airflow—kg/sec (lb/sec) ● Suppression	ATAT-1 287 (633) Plug and chutes	ATAT-1 287 (633) Plug and chutes	ATAT-1 287 (633) Plug and chutes
Weights ● OEW relative to 969-512B—kg (lb)	721.21 (1,590)	3,048.14 (6,720)	0
Range ● Supersonic cruise, km (nmi) range relative to 969-512B ● Supersonic cruise altitude, m (ft)	-53.74 (-29) 19,507.2 (64,000)	-359.52 (-194) 19,630 (64,500)	0 19,507.2 (64,000)
Cruise performance ● Supersonic R.F. km (nmi) at M = 2.7 ● L/D (L/D)_{max} ● SFC, kg/hr per kg (lb/hr per lb)	15,268.35 (8,239) 8.29/8.63 1.559	14,908.83 (8,045) 8.17/8.5 1.573	15,268.35 (8,239) 8.29/8.63 1.559
Takeoff ● FAR field length, m (ft) (std + 15 K) ● Liftoff speed, m/sec (KEAS) ● Sideline noise, EPNdB ● Community noise, 6.49 km (3.5 nmi) from brake release EPNdB	3,780 (12,400) 101 (197) 117 107	3,780 (12,400) 101 (197) 117 107	3,780 (12,400) 101 (197) 117 108
Approach ● Approach speed, m/sec (KEAS) ● Wet FAR field length, m (ft) ● L/D ● Approach noise, 1.85 km (1 nmi) threshold, EPNdB	83 (162) 3,139.44 (10,300) 4.43 117	83 (162) 3,139.44 (10,300) 4.9 114.5	84 (163.5) 3,200.4 (10,500) 4.23 117.5

*Note: 969-512 and 969-513 are referenced to the 969-512B.

TABLE 1.8.—TAIL SIZE STUDY—MODEL 963 512

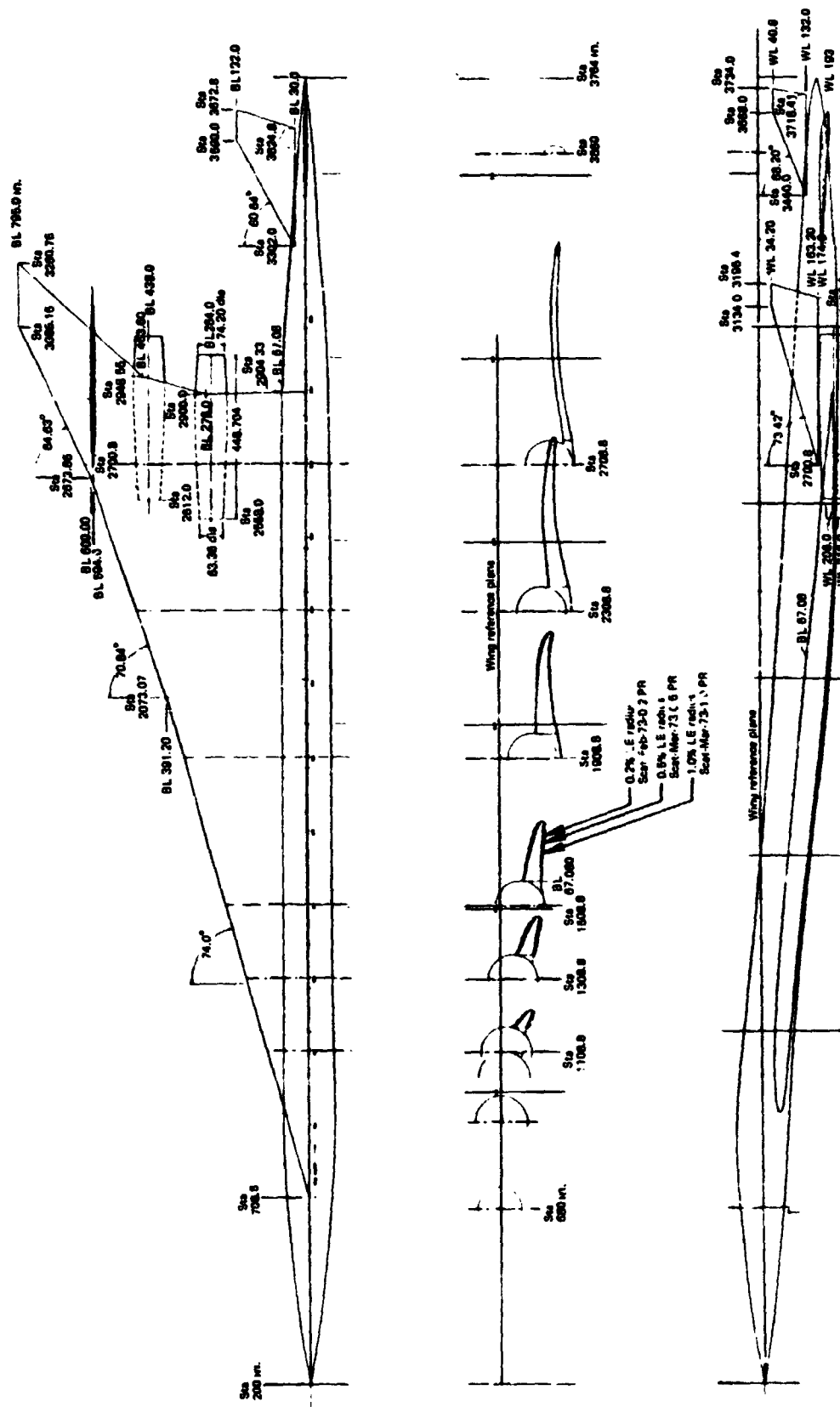
	Large tail	Δ OEW kg (lb)	Small tail
Cg limits			
Aft T.O. and landing—%MAC	51.6		49.8
Fwd T.O.—%MAC	49.4		47.5
Fwd landing—%MAC	48.3		46.5
Horizontal tail		-885 (-1,950)	
Area—m ² (ft ²)	55.74 (600)		44.50 (479)
$\bar{V}$	0.0545		0.0435
Tail arm—m (in.) (from 50% c _w)	26.97 (1,062)		26.97 (1,062)
Vertical tail		+32 (+70)	
Area—m ² (ft ²)	41.71 (449)		42.27 (455)
$\bar{V}$	0.028		0.029
Tail arm—m (in.) (from aft T.O. limit)	24.82 (977)		25.10 (988)
Landing gear		+218 (+480)	
Landing gear length—m (in.)			+0.11 (+4.3)
Body		+807 (+1,780)	
Body length—m (in.) (balance airplane with forebody length)			+1.27 (+50)
Net weight change		+172 kg (+380 lb)	

TABLE 1-9.—WEIGHT DIFFERENCES, MODEL 969-512 TO MODEL 969-512B

	OEW	
	kg	(lb)
Model 969-512		
Revise high speed roll control	161,792	(356,690)
Surfaces and actuation	-177	(-390)
Move cg limits 1.5% aft and move body 0.44 m (17.5 in.) on wing for rebalance		
Move main gear aft 0.89 m (35 in.) and decrease main gear length 0.30 m (12 in.)	-503	(-1,110)
Increase nose gear length 0.25 m (10 in.)	+70	(+155)
Delete 0.76 m (30 in.) of forward body system runs due to weight duplication	-156	(-345)
Add alpha limiter previously omitted	+45	(+100)
Net weight change	[-721]	[(-1,590)]
Model 696-512B	161,071	(355,100)

TABLE 1-10.—GROUP WEIGHT AND BALANCE STATEMENT, MODEL 969-512B

Group	Wt (kg)	Arm (m)	Wt (lb)	Arm (in.)
Wing	42,710	66.14	94,160	2,604
Horizontal tail	2,962	92.02	6,530	3,623
Vertical tail (body and wing mtd)	2,268	86.51	5,000	3,406
Body	25,465	53.77	56,140	2,117
Main gear	16,928	64.72	37,320	2,548
Nose gear	1,705	29.92	3,760	1,178
Nacelle	8,655	74.90	19,080	2,949
Total structure	100,693	64.13	221,990	2,525.0
Engine (incl T/R, S/S and nozzle)	20,502	78.13	45,200	3,076
Engine accessories	612	74.78	1,350	2,944
Engine controls	354	58.62	780	2,308
Starting system	136	74.14	300	2,919
Fuel system	3,806	63.37	8,390	2,495
Total propulsion	25,410	75.55	56,020	2,974.3
Instruments	846	43.43	1,865	1,710
Flight controls	6,668	68.05	14,700	2,679
Hydraulics	2,628	72.49	5,795	2,854
Electrical	2,341	53.14	5,160	2,092
Electronics	1,309	32.56	2,885	1,282
Furnishings	8,623	46.15	19,010	1,817
ECS	3,824	61.98	8,430	2,440
Anti-icing	61	14.17	135	558
APU	113	75.64	250	2,978
Insulation	1,315	48.59	2,900	1,913
Total systems and equipment	27,728	56.13	61,130	2,209.7
Options	1,134	63.27	2,500	2,491
Manufacturer's empty weight	154,965	64.56	341,640	2,542.0
Standard items	3,719	55.70	8,200	2,193
Operational items	2,386	43.59	5,260	1,716
Operational empty weight	161,070	64.05	355,100	2,521.7
Payload	22,183	47.80	48,906	1,882
Zero fuel weight	183,253	62.09	404,006	2,444.3



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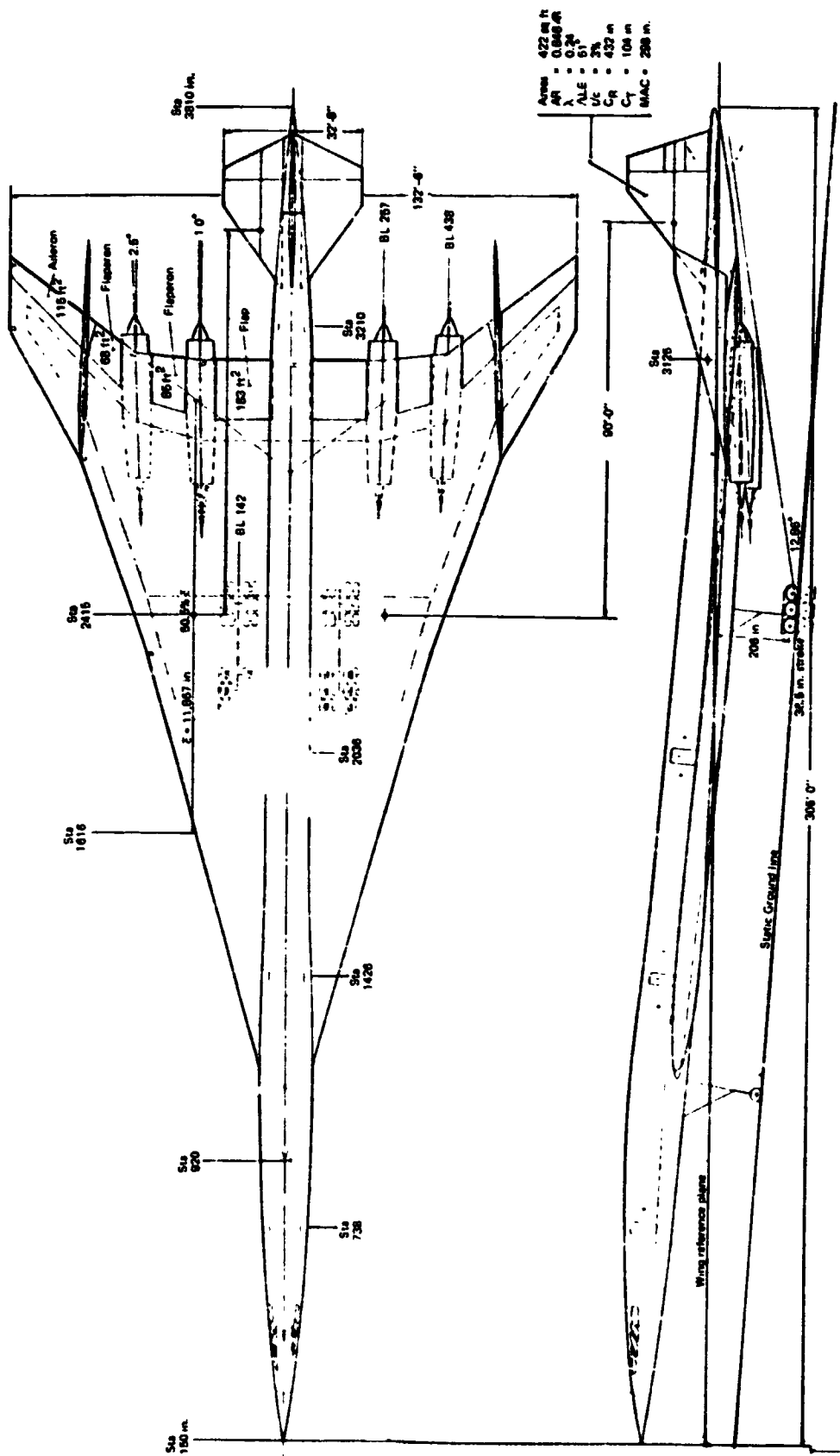


FIGURE 1-2.--ALTERNATE CONFIGURATION, ENGINES OUTBD, MODEL 969-512

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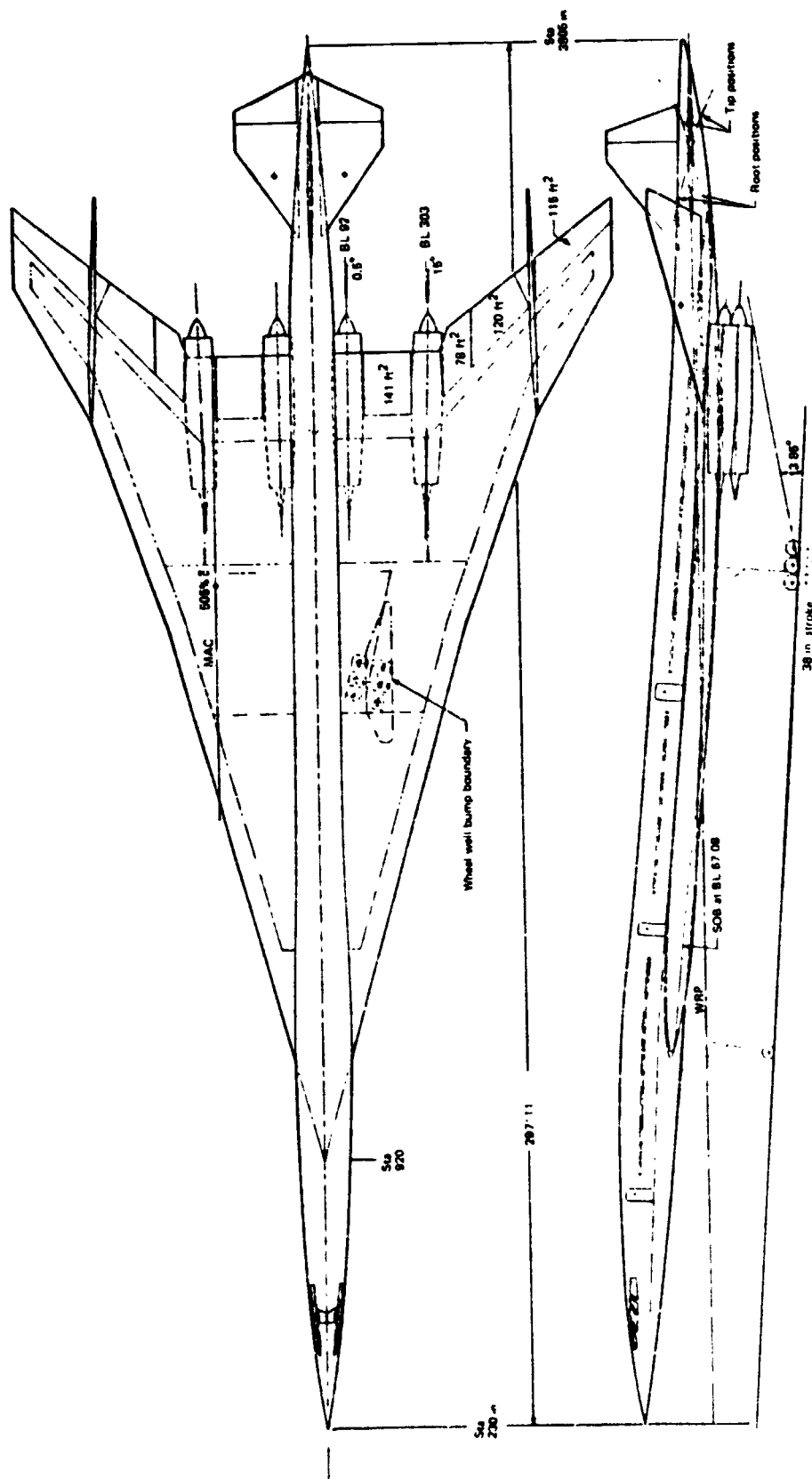


FIGURE 1-3.-ALTERNATE CONFIGURATION, ENGINES INBD, MODEL 969-513

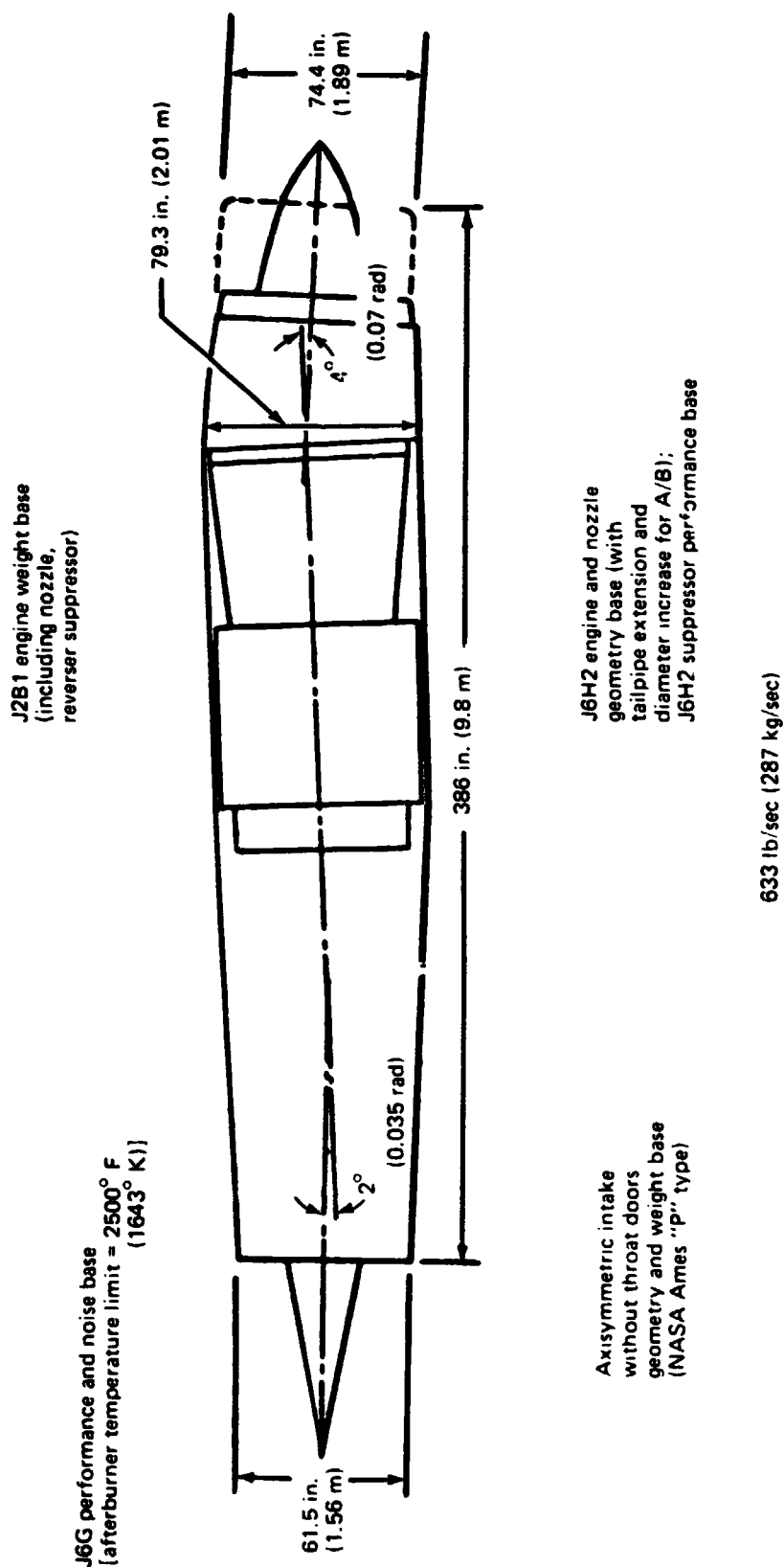


FIGURE 1-4.—ATAT-1 AFTERBURNING TURBOJET POD GEOMETRY

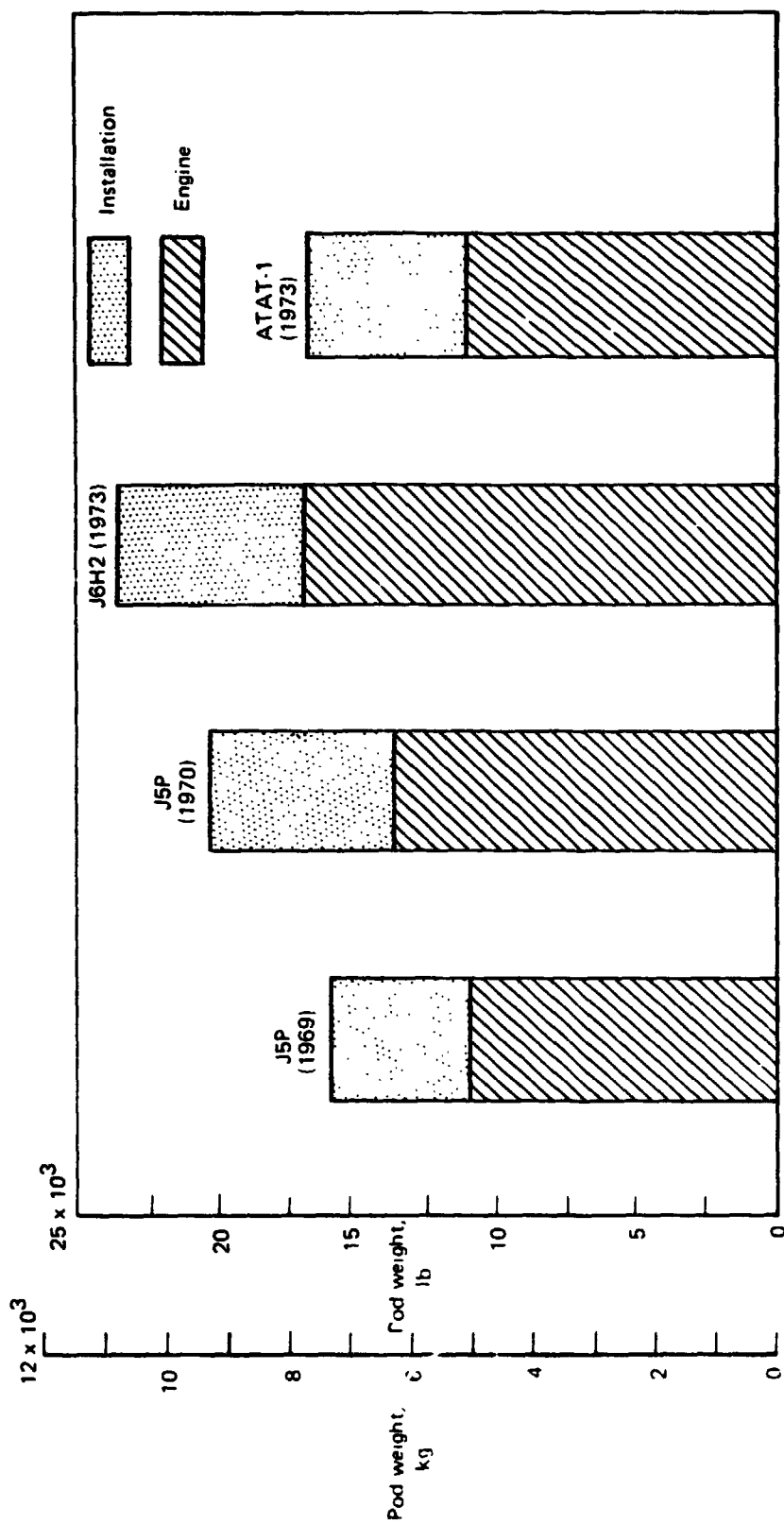


FIGURE 1-5.—POD WEIGHT COMPARISON

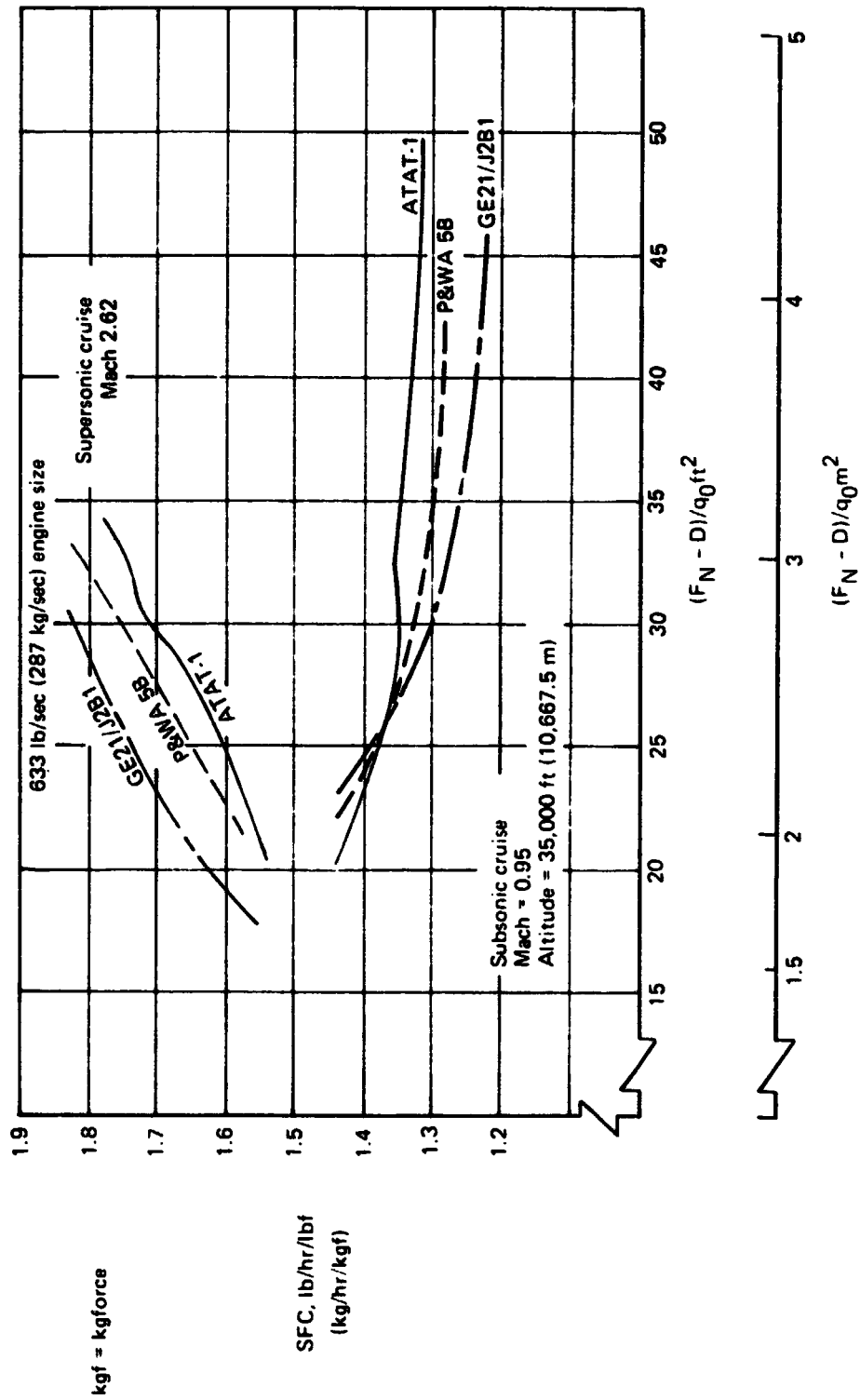


FIGURE 1-6.—CRUISE PERFORMANCE COMPARISON

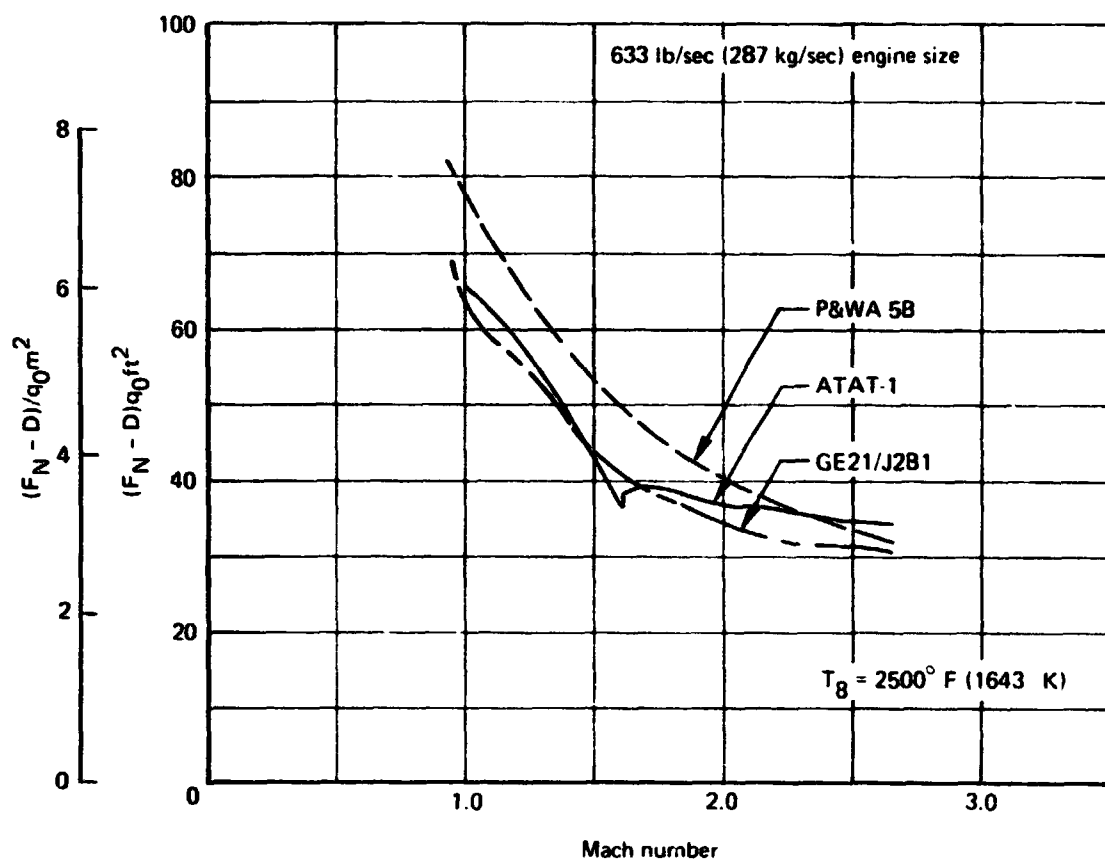


FIGURE 1-7.—CLIMB PERFORMANCE COMPARISON

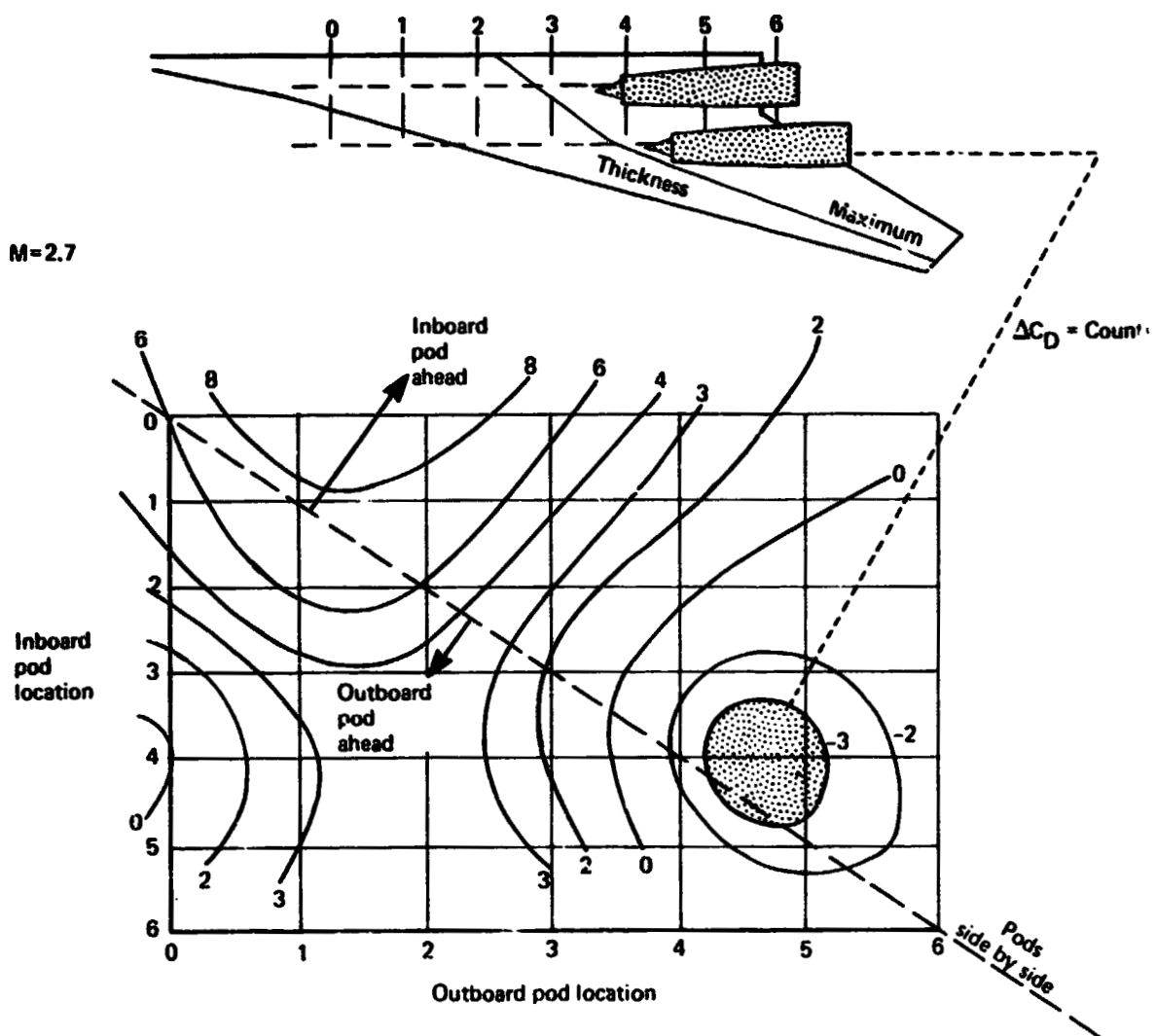


FIGURE 1-8.—EFFECT OF NACELLE LOCATION ON WAVE DRAG

Note:
 Surface noise levels in dB's
 ATAT-1 afterburning turbo jet engine
 max SLS power, $T_{T_{exit}} = (1645 \text{ K}) 2500 \text{ F}^\circ$
 Suppressors not deployed

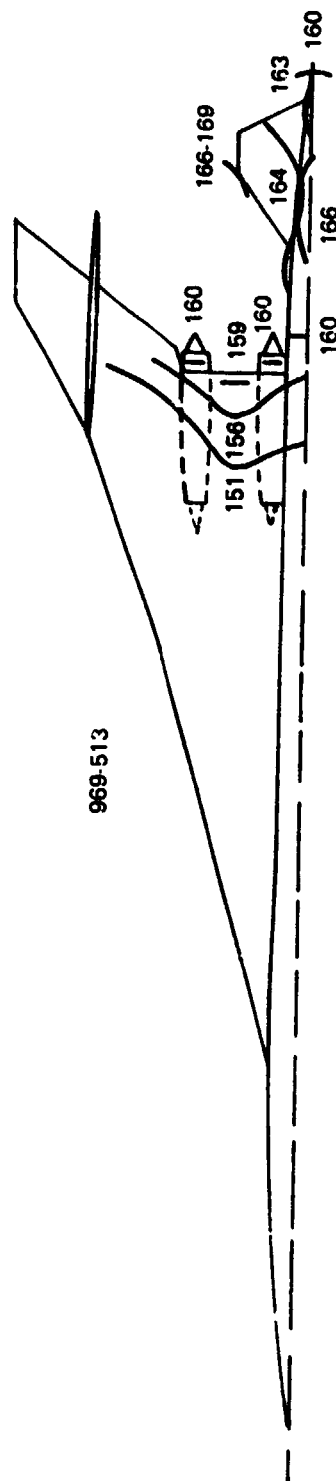
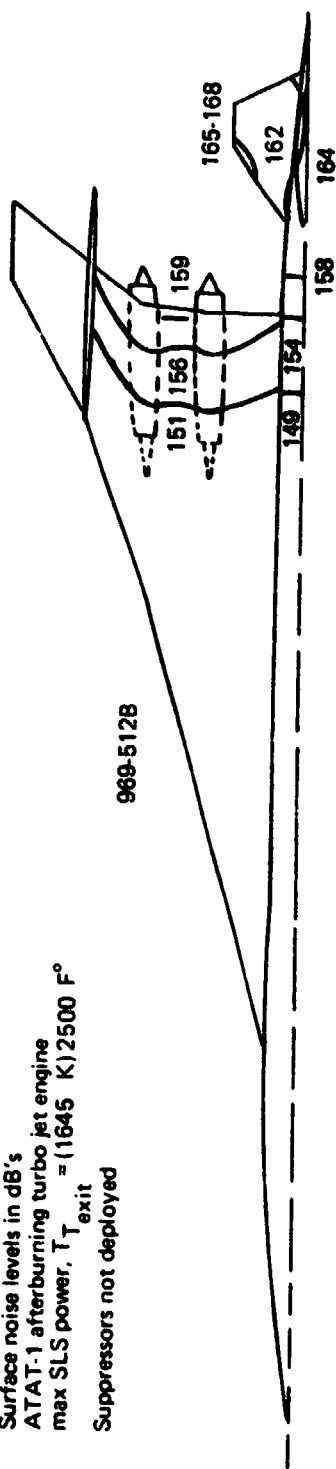


FIGURE 1.9.—ESTIMATED NOISE LEVELS ON SURFACES OF TWO ARROW WING CONFIGURATIONS

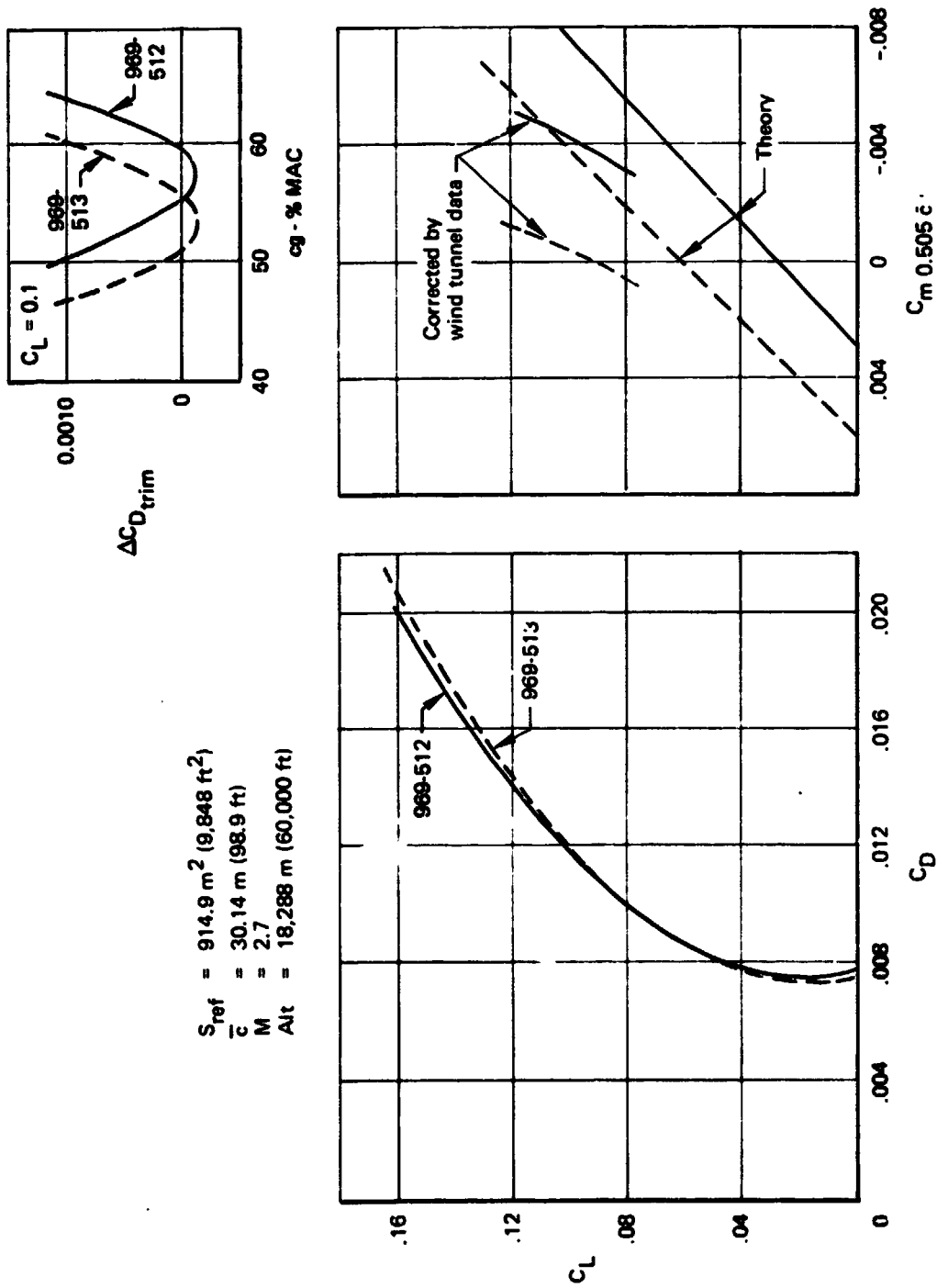


FIGURE 1-10.—DRAG AND MOMENT COMPARISON

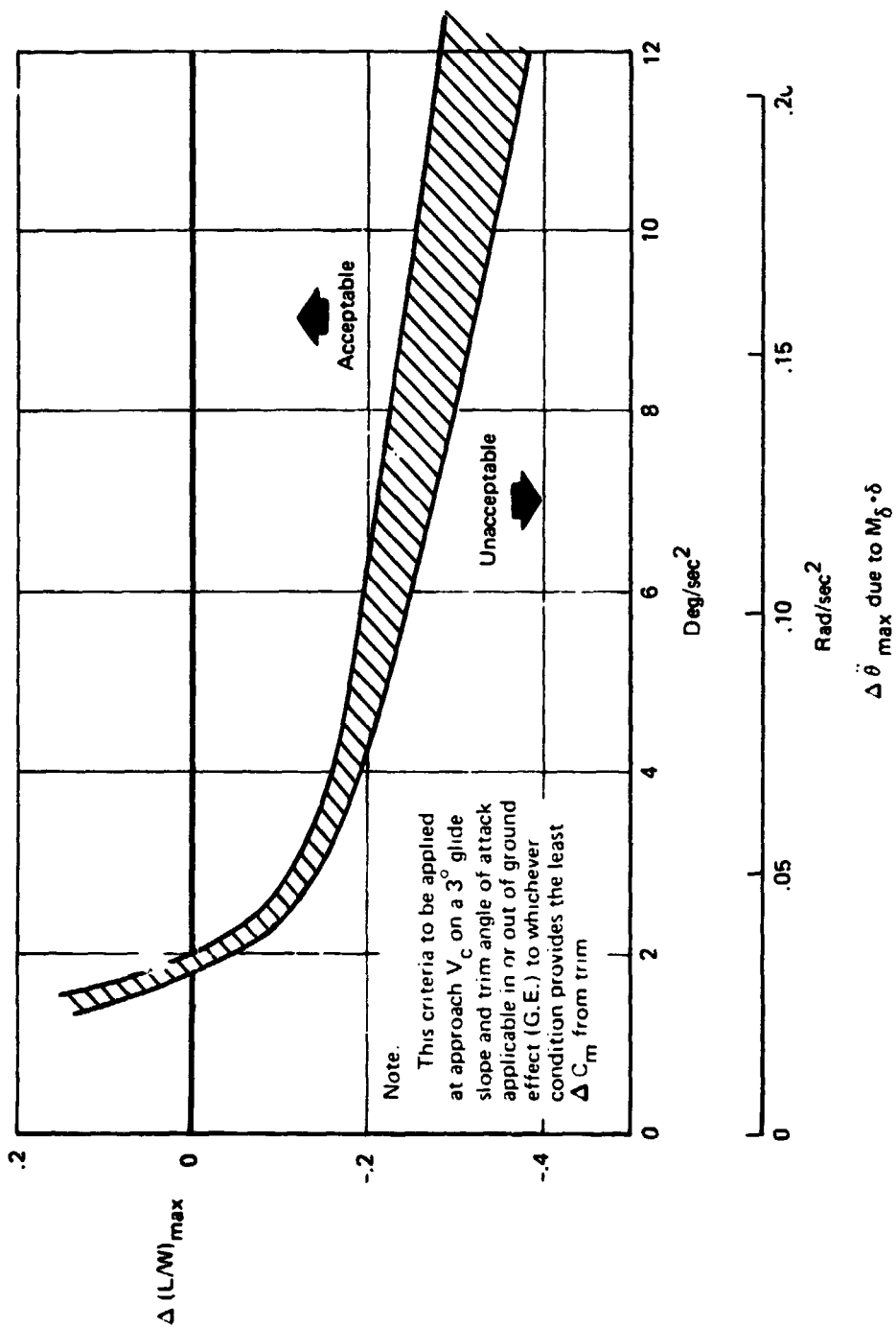


FIGURE 1-11.—LANDING APPROACH PITCH ACCELERATION CRITERIA

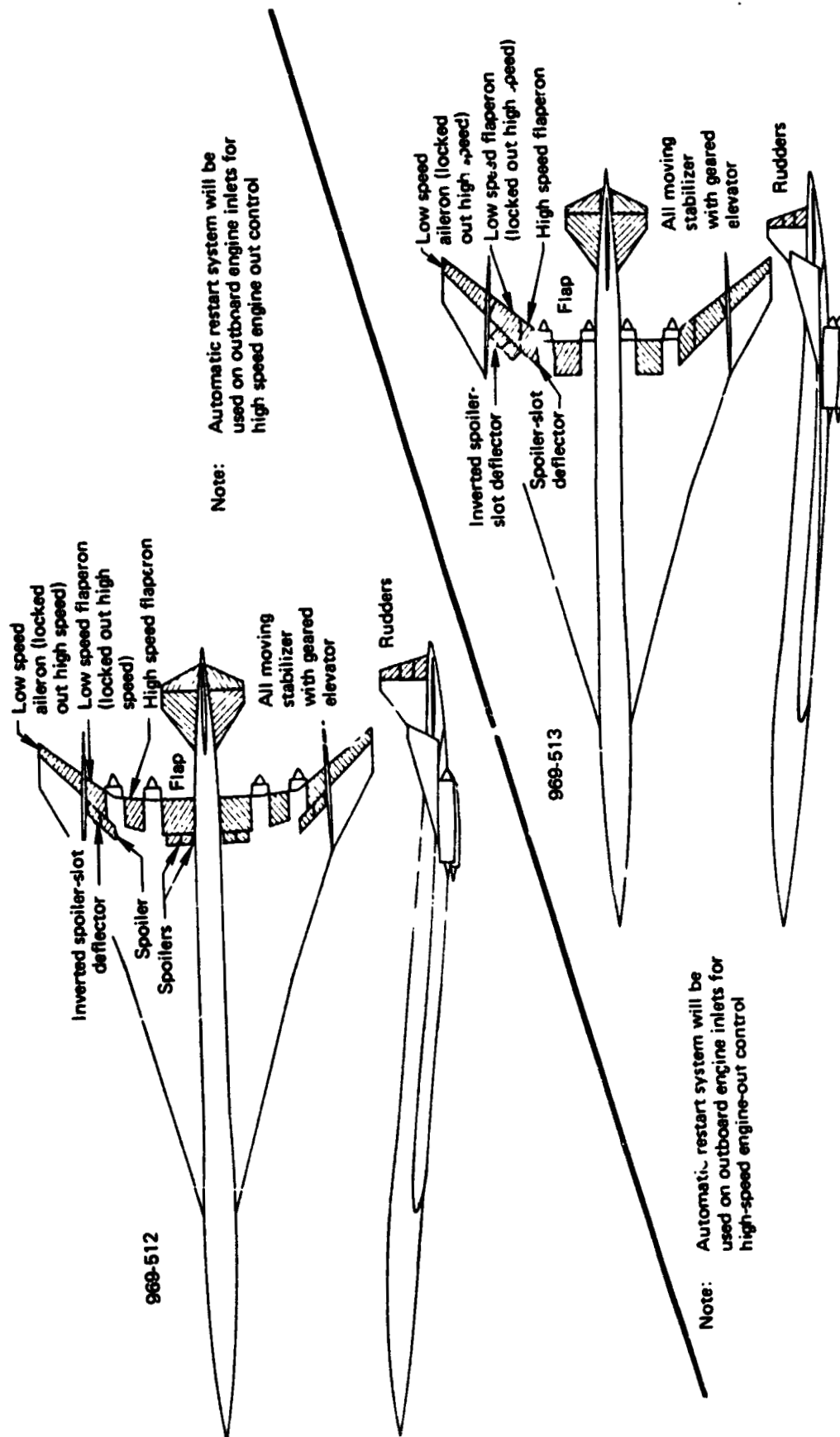


FIGURE 1-12. -969-512/-513 CONFIGURATION COMPARISON

Note: l_H measured from $0.50 \bar{c}_{ref}$ to horizontal tail quarter chord

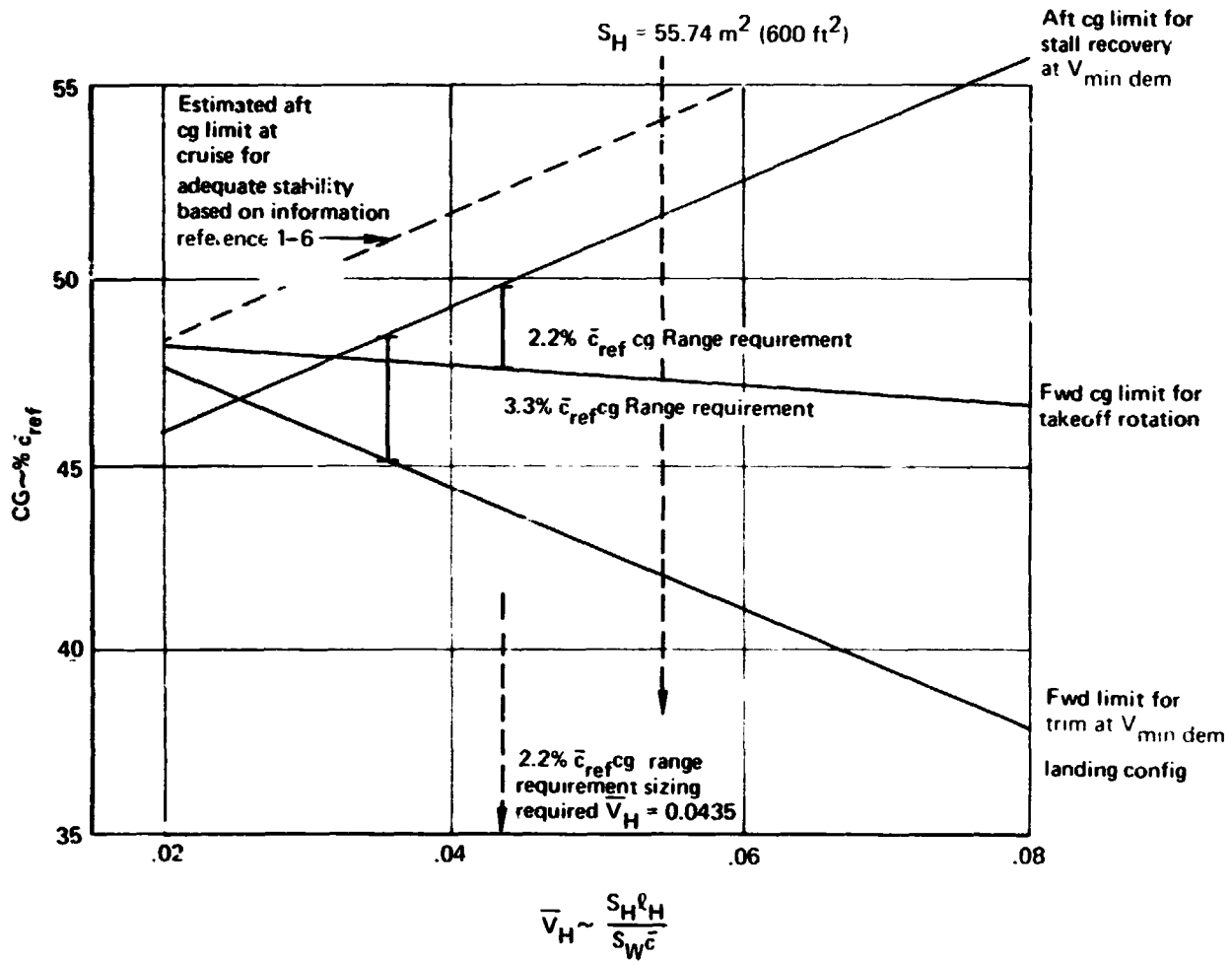


FIGURE 1-13.—HORIZONTAL TAIL SIZING CHART
(969-512/-513)

Notes:

- 1) Chart shows body-mounted vertical tail volume coefficient variation with inboard and outboard engine location
- 2) Chart sizes body-mounted vertical to meet maximum side-slip requirements during engine seizure, inlet unstart failure condition at $M = 2.7$ climb
- 3) Tail moment arm to be measured from the aft cg limit
- 4) Sizing of the 969-512 body-mounted vertical shown as an example

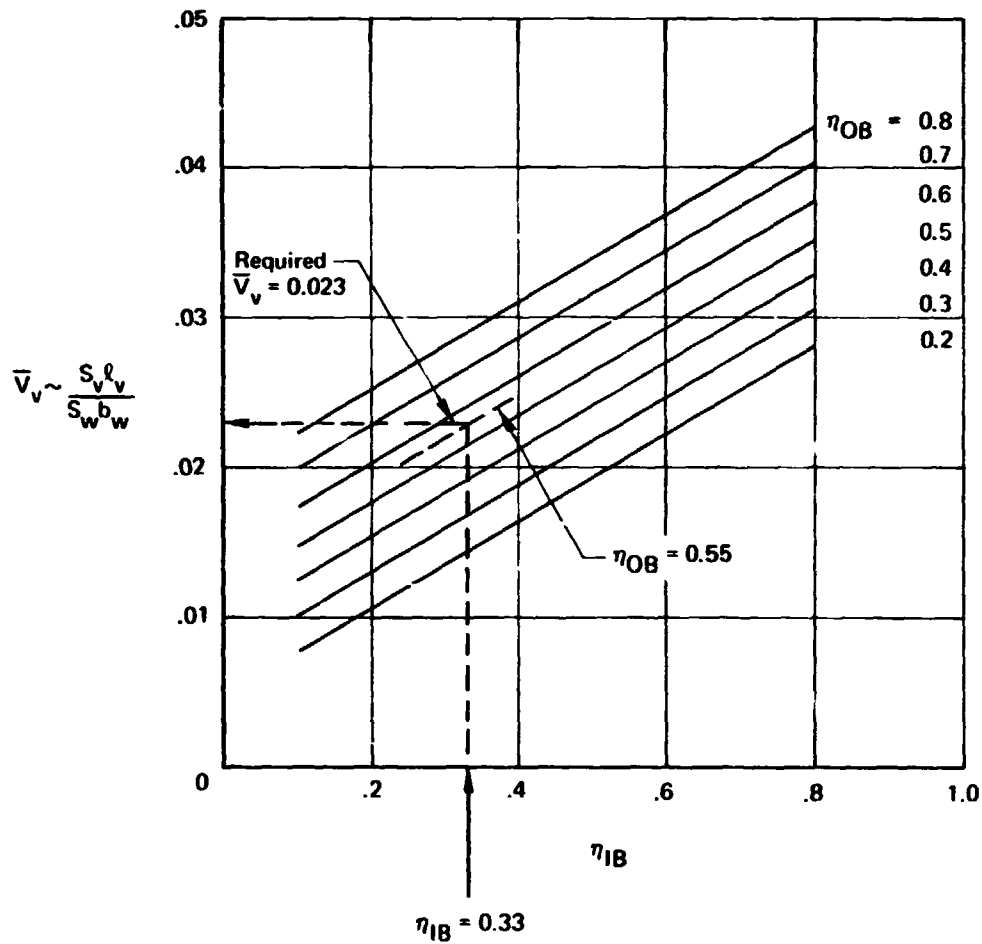


FIGURE 1-14.—VERTICAL TAIL SIZING CHART (969-512/-513)

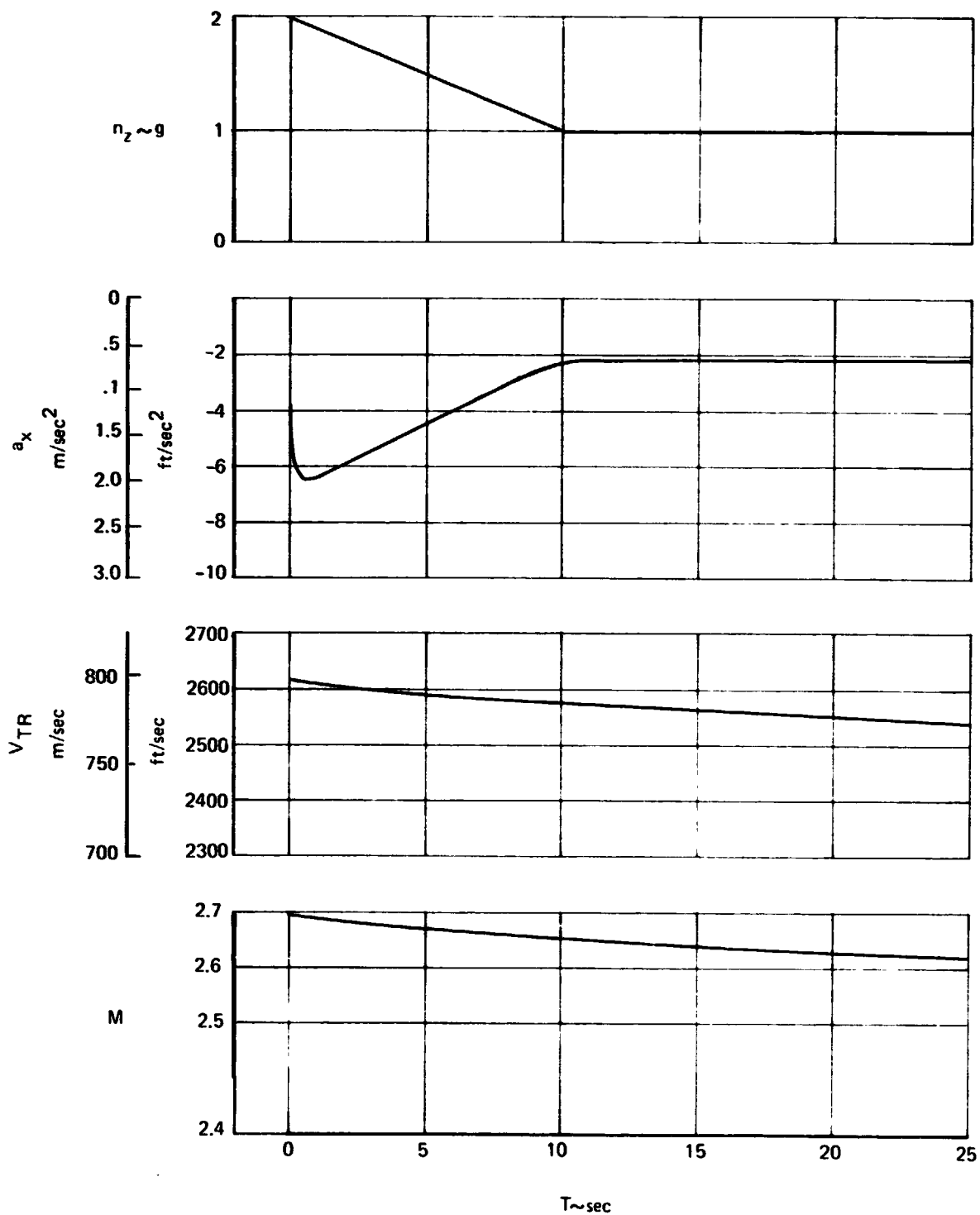


FIGURE 1-16.—TIME HISTORY TWO-ENGINE FLAME-OUT
 $M = 2.7$ WITHOUT AUTOMATIC RESTART

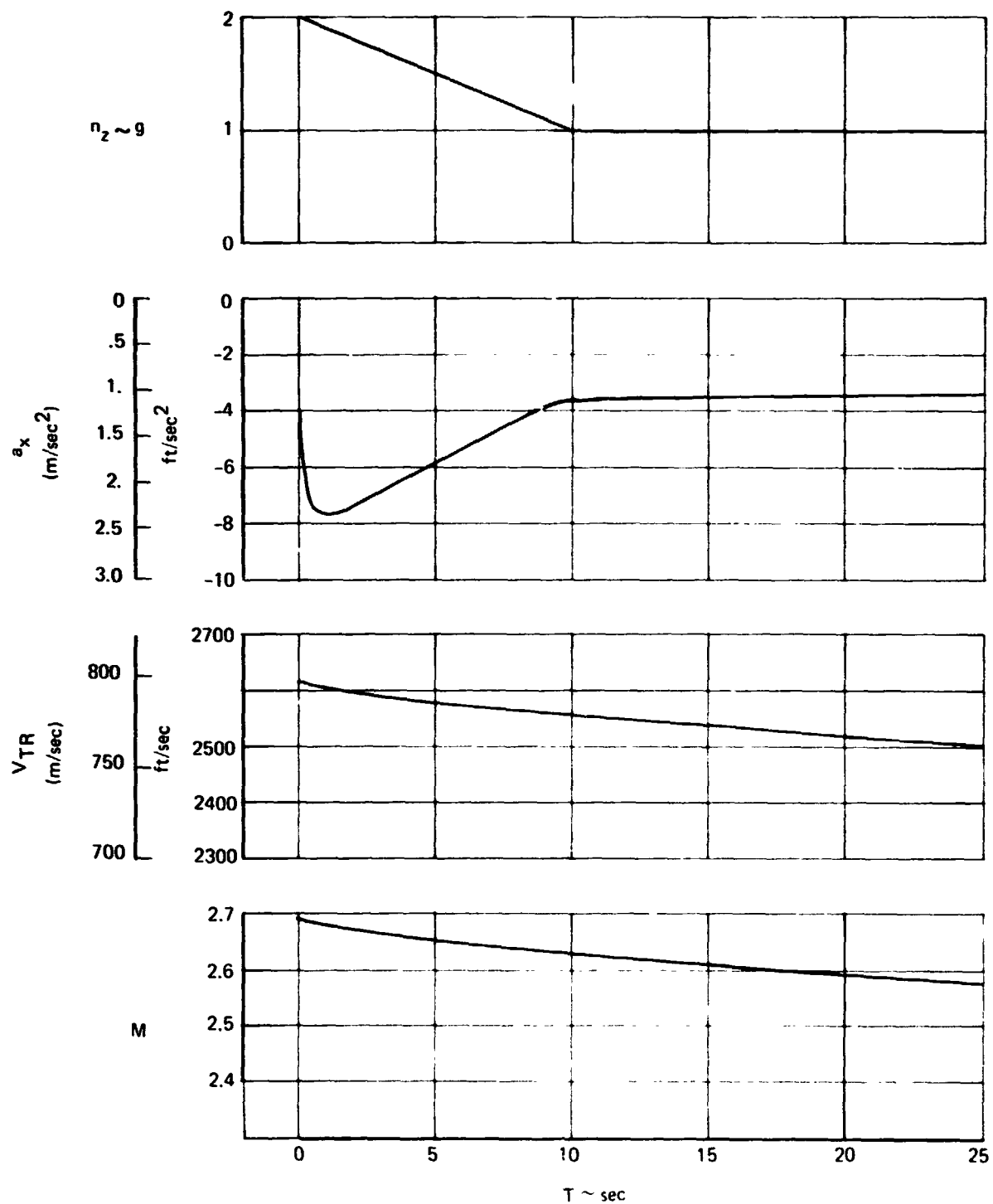


FIGURE 1-17.—TIME HISTORY TWO-ENGINE FLAME-OUT $M = 2.7$
WITH AUTOMATIC RESTART

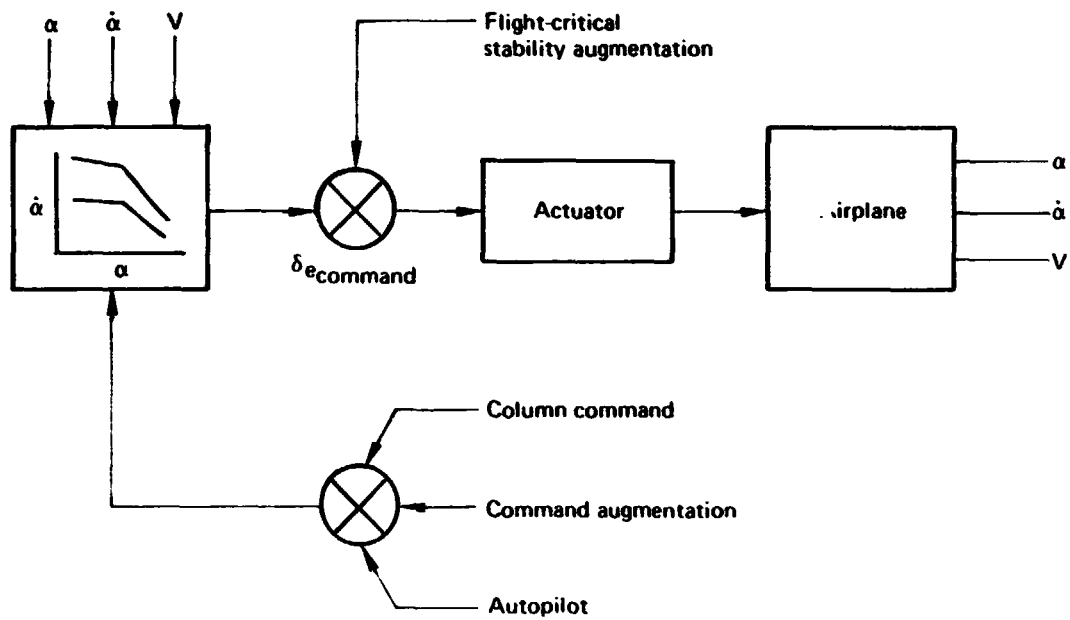


FIGURE 1-18. ALPHA LIMITER FUNCTIONAL DIAGRAM

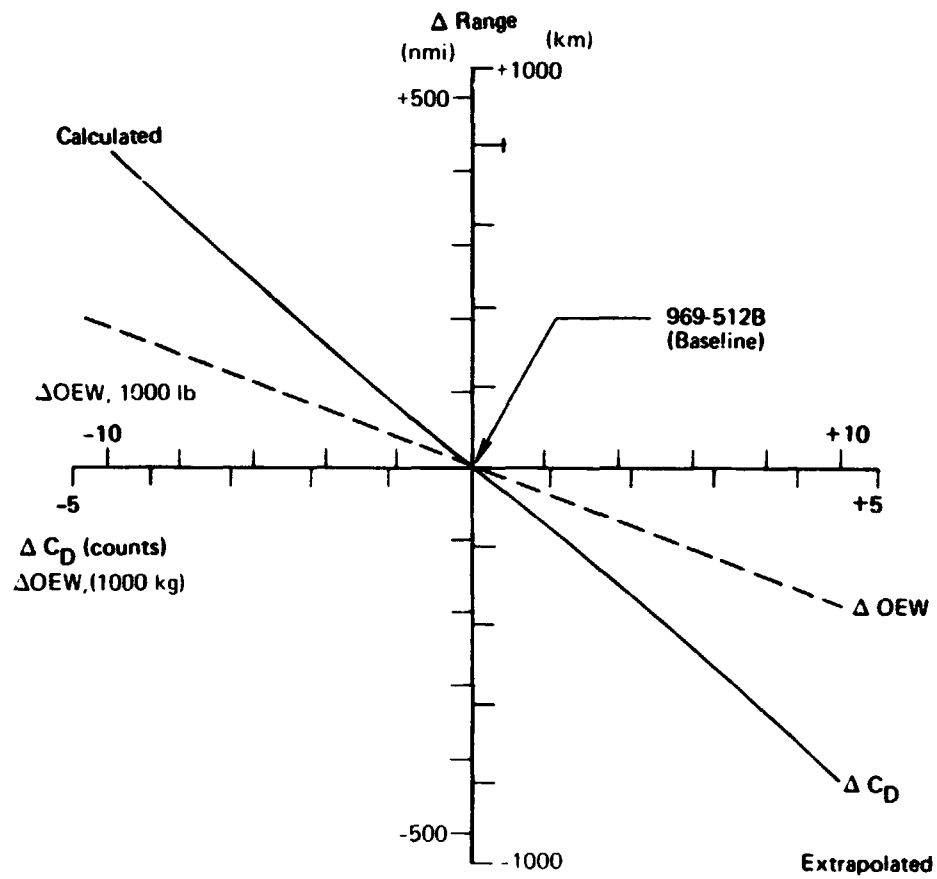


FIGURE 1-19. - RANGE SENSITIVITY TO OEW AND SUPERSONIC DRAG

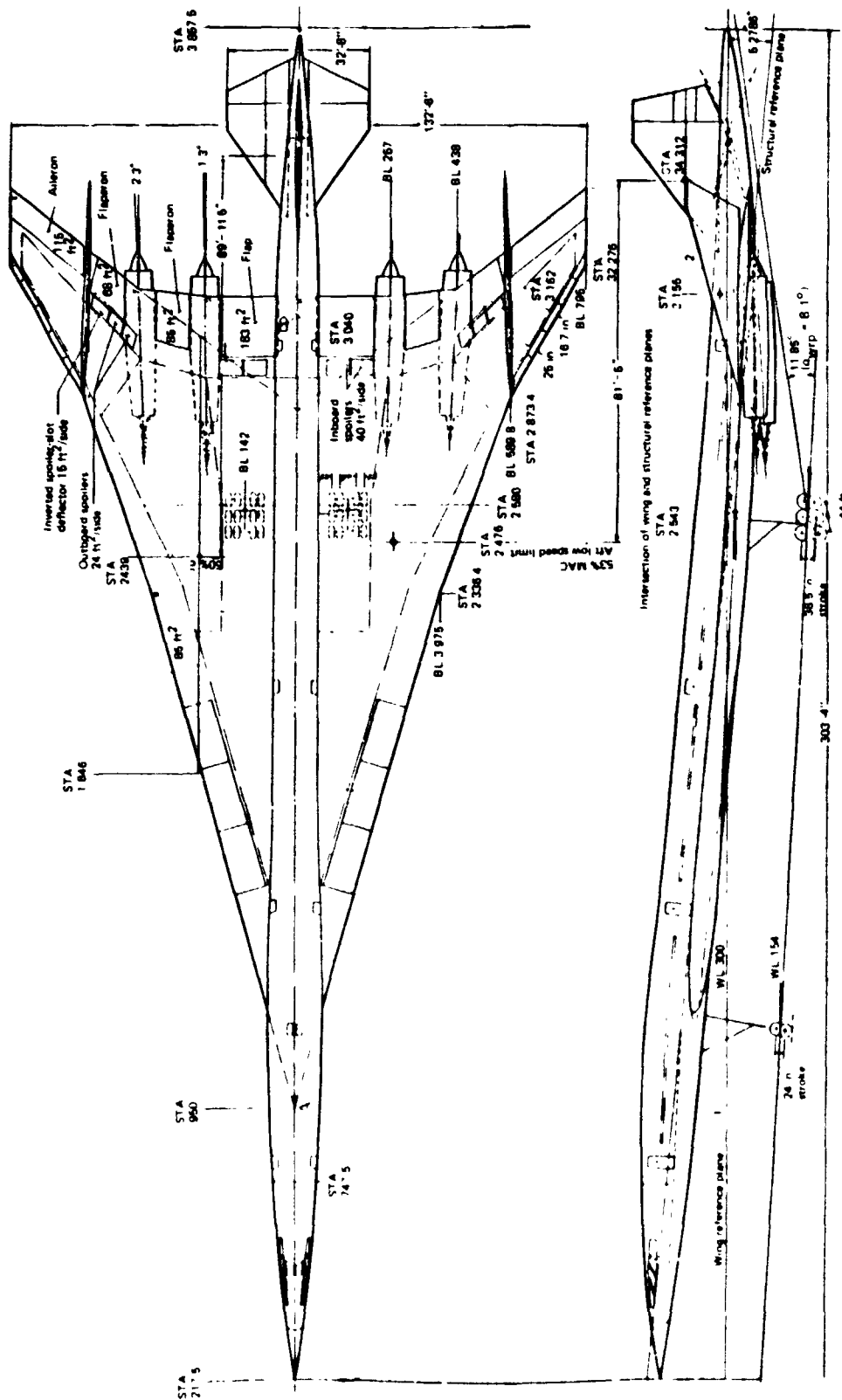


FIGURE 1-20.-CONFIGURATION FOR STRUCTURAL ANALYSIS, MODEL 969-512B

Geometry		Wing	Wing Vert. stabilizer	Vertical stabilizer	Horizontal stabilizer
Area	sq ft	9,848*	287/side	449	600 exposed
Aspect ratio	λ	1.78	0.493	0.848	1.32
Taper ratio	λ		0.135	0.24	0.247
Sweep at LE	deg	74/70.5/60	74.5	51°	54°
Incidence	deg	—	—	—	-15/30,+15/25
Dihedral	deg	—	—	—	0
Root t/c	%	—	3	3	3
Tip t/c	%	—	3	3	3
Root chord	in.	1881.1	510	445	414
Tip chord	in.	204	69	107	102
MAC	in.	1187	346	311	289
Span	in.	1590	143	—	338
Tail arm	in.	—	697	977	1062
Tail vol coeff	$\bar{v}$	—	0.013	0.028	0.0545

* Reference area. Total wing area = 11,244 sq ft

Gross weight 750,000 lb

Body	Length in.	Max dia in.	Accommodation	
	3640	102.0	234 pass.	4/5 AB
Powerplants	Number	Type	Airflow	Inlet
	4	ATAT-1	633 lb/sec	Axisym
Landing gear	Nose	Main	Loc % MAC	
	2 34X16	24 40 7X14	57.7 (pivot)	
Fuel capacity lb	Wing	Body	Total	cg % C _{ref}
	317,400	72,000	389,000	
cg limits		Takeoff	Cruise	Landing
S _{AC}	Fwd			49.7
	Aft		55.5	53.0

FIGURE 1-20.—CONFIGURATION FOR STRUCTURAL ANALYSIS, MODEL 969-512B (CONCLUDED)

- Note. 1) ℓ_H measured from 50% $\bar{c}_{ref}$ to be 1079.5 inches.
 2) Main gear truck pivot to be located such that its projection onto reference plane is 4.7% aft of the aft

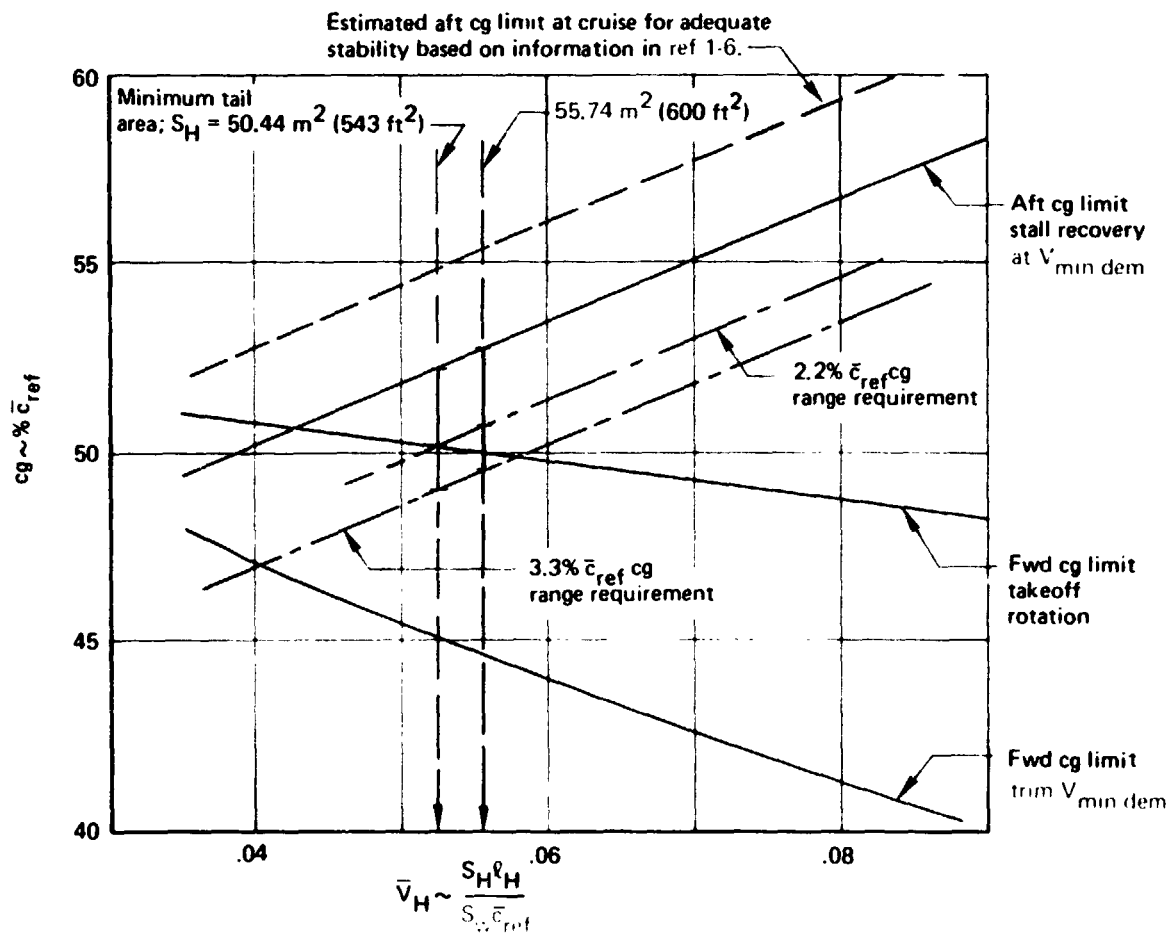


FIGURE 1-21. - HORIZONTAL TAIL SIZING CHART
 (969-512B)

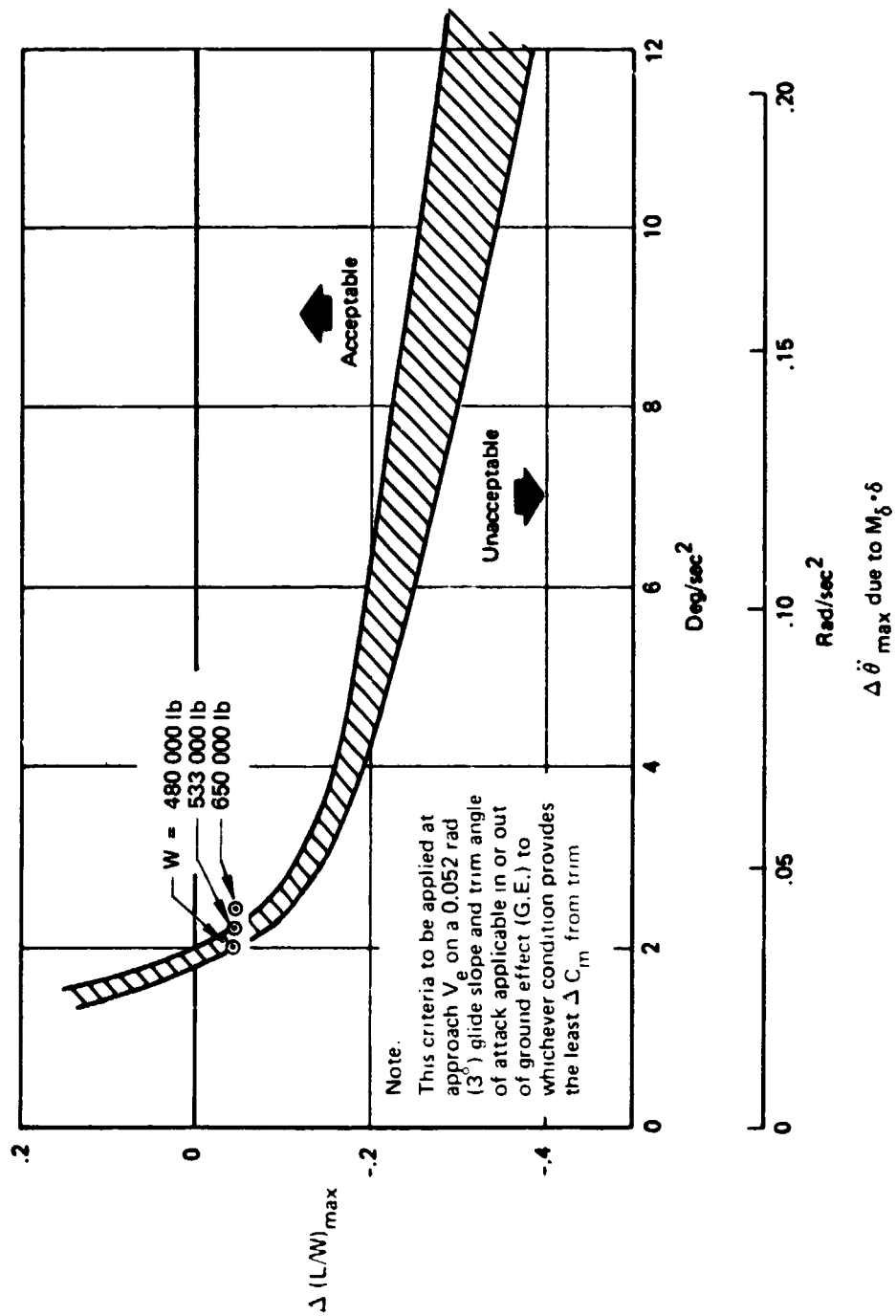


FIGURE 1-22. —LANDING APPROACH PITCH ACCELERATION EVALUATION, 969-512B

- Note.
- 1) Curves based on Langley flaps down and Ames flaps up low speed data
 - 2) Corrections made for planform and flap geometry differences
 - 3) Airplane weight $W = (226,796 \text{ kg}) 500,000 \text{ lb}$

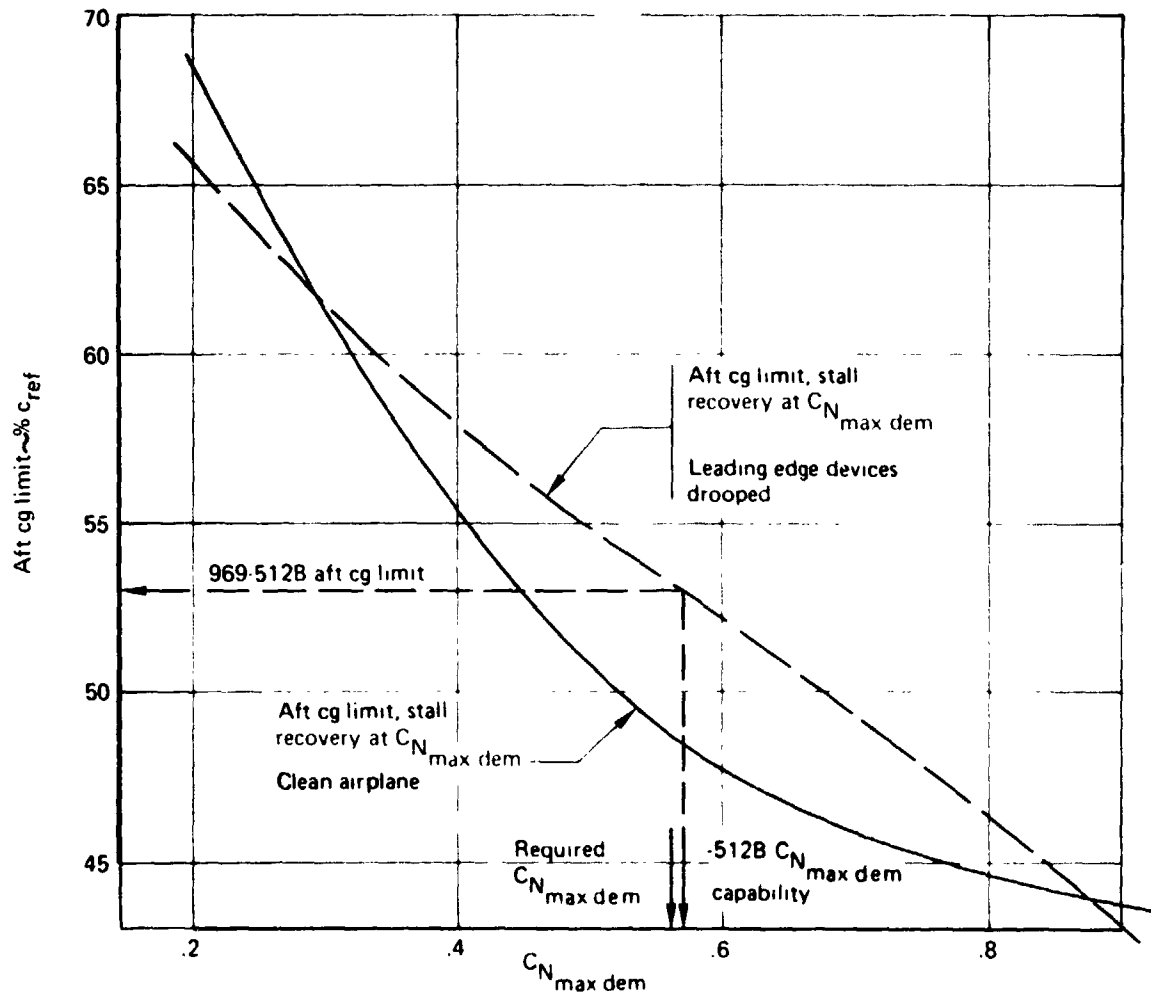


FIGURE 1-23 -EFFECTS OF MAXIMUM NORMAL FORCE COEFFICIENT ON LOW SPEED AFT CG LIMIT

Note: All variables are
perturbation variables

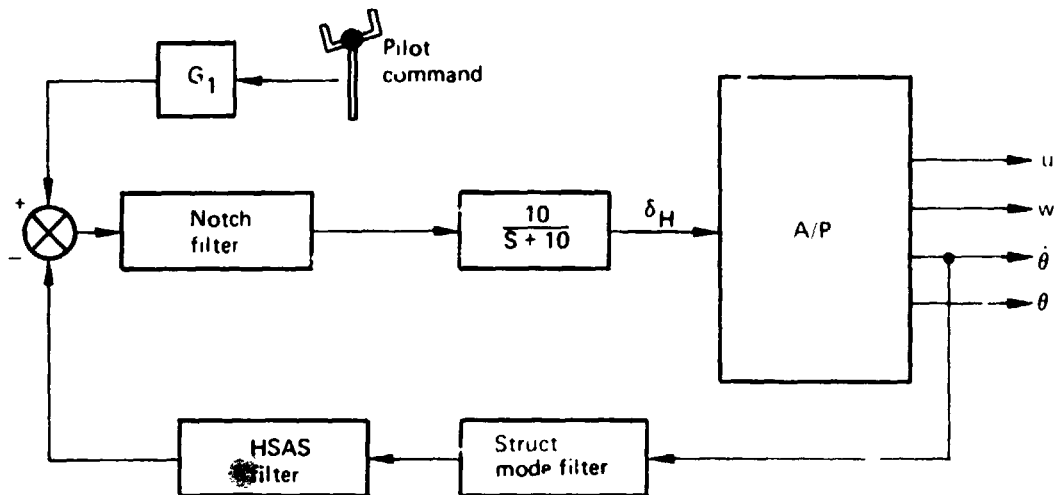


FIGURE 1-24. --LONGITUDINAL HSAS FUNCTIONAL DIAGRAM

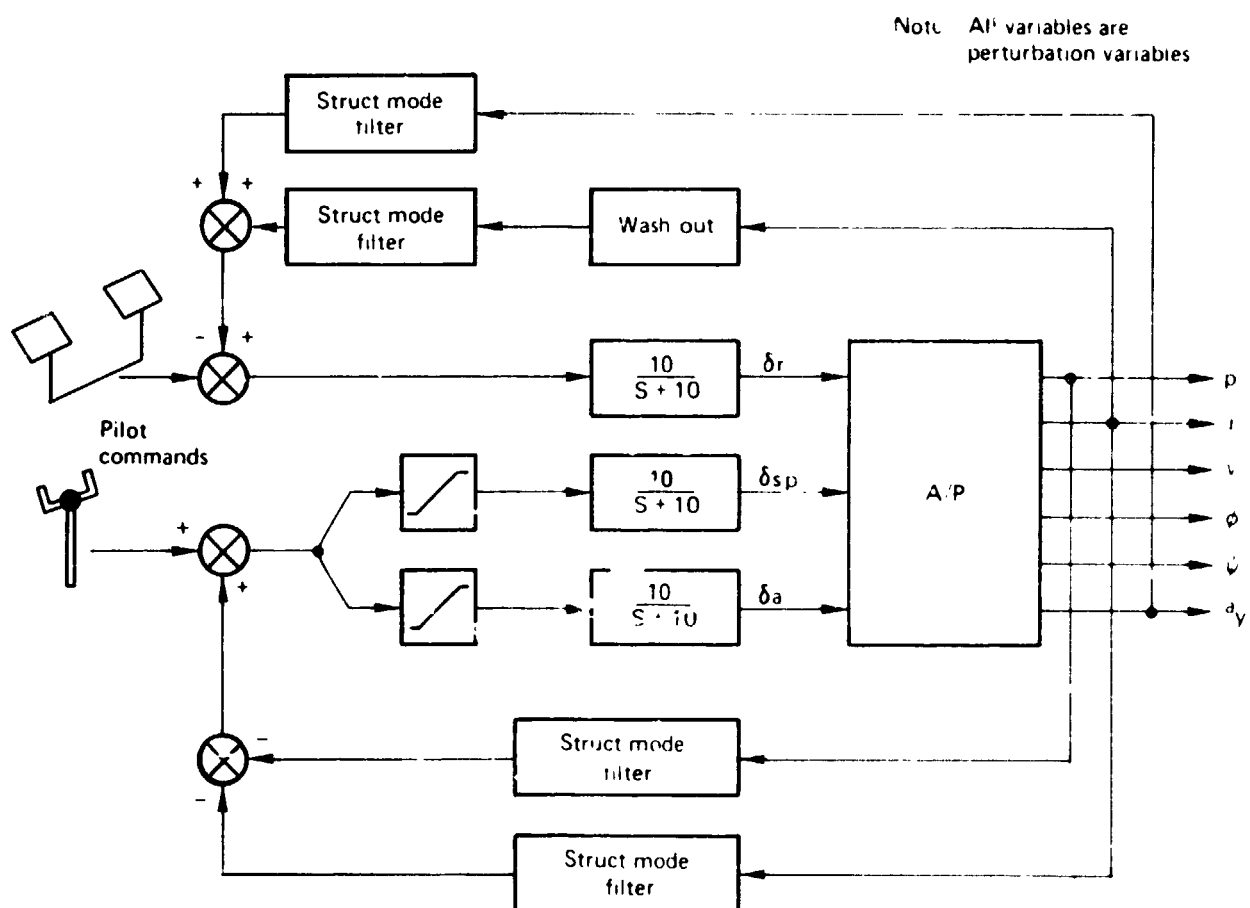


FIGURE I-25. LATERAL-DIRECTIONAL HSAS FUNCTIONAL DIAGRAM

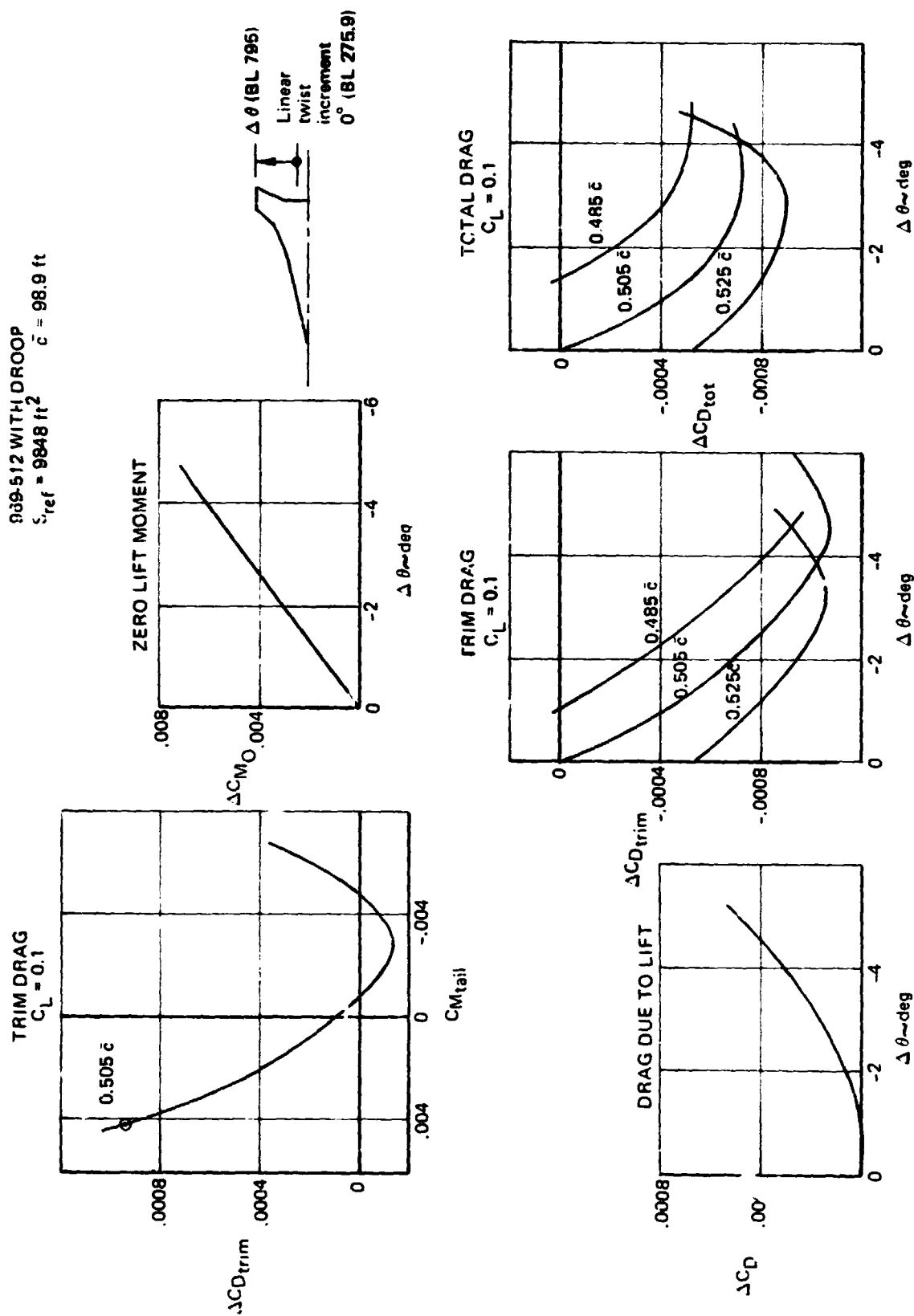
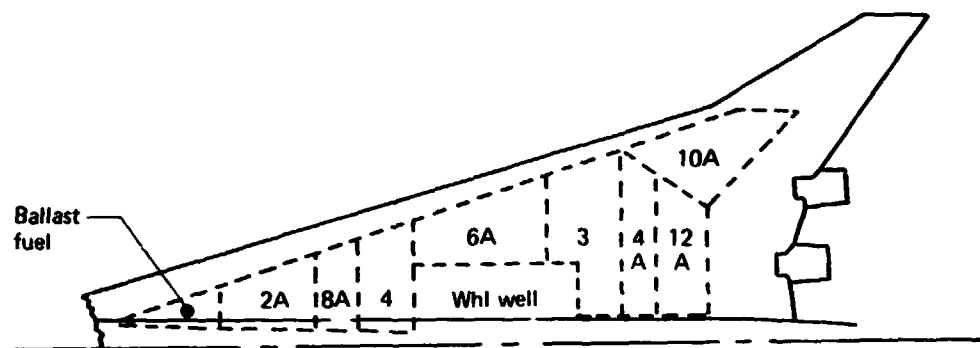


FIGURE 1-26. EFFECT OF OUTBOARD WING TWIST-969-512 WITH DROOP



Tank no.		Capacity/airplane		cg body station	
Left	Right	kg	lb	m	in.
1	4	27,488	60,600	53.72	2,115
2	3	31,207	68,800	66.60	2,622
1A	2A	33,747	74,400	47.07	1,853
3A	4A	17,373	38,300	69.16	2,723
5A	6A	40,279	88,800	60.25	2,372
7A	8A	12,156	26,800	51.05	2,010
9A	10A	15,876	35,000	73.81	2,906
11A	12A	20,366	44,900	71.76	2,825
Ballast		10,614	23,400	41.66	1,640
Total capacity (excluding ballast)		198,492	437,600	60.58	2,385.2

FIGURE 1-27. -FUEL TANK LAYOUT-MODEL 969-336C

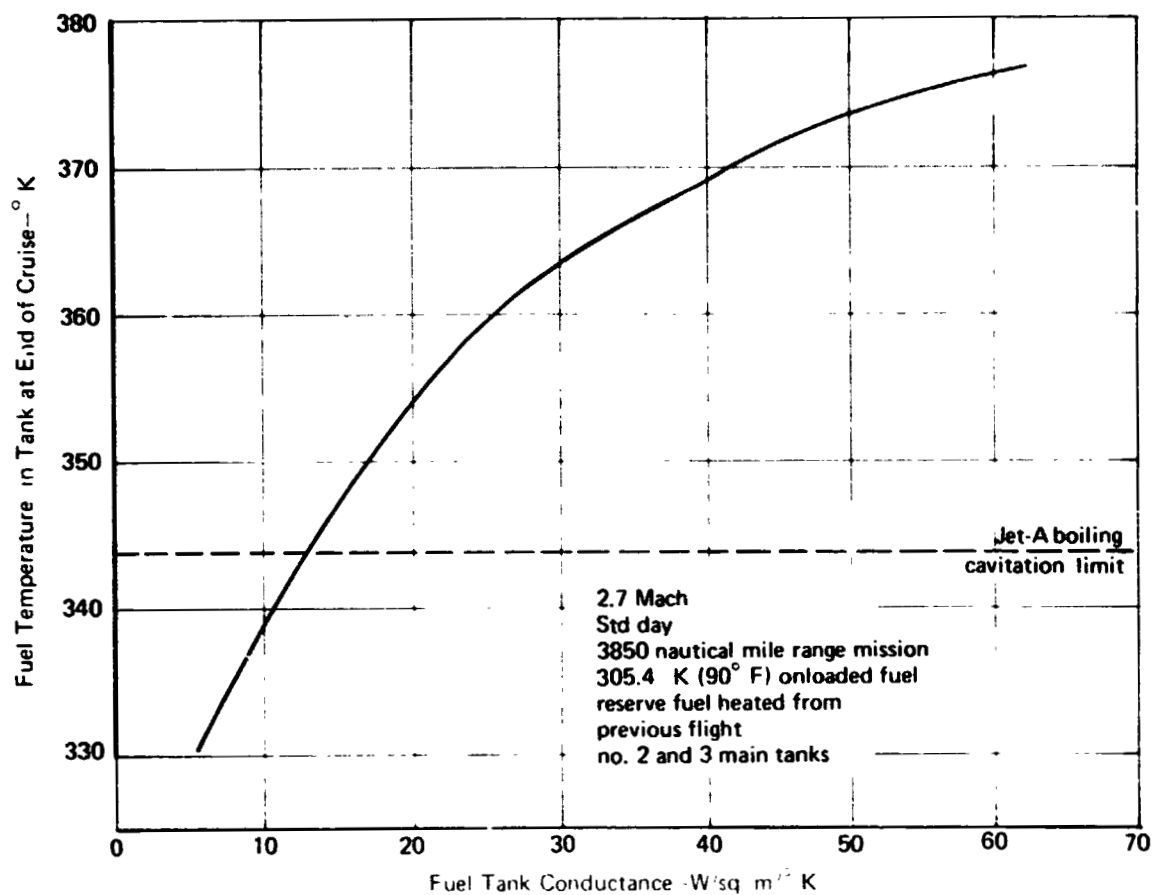
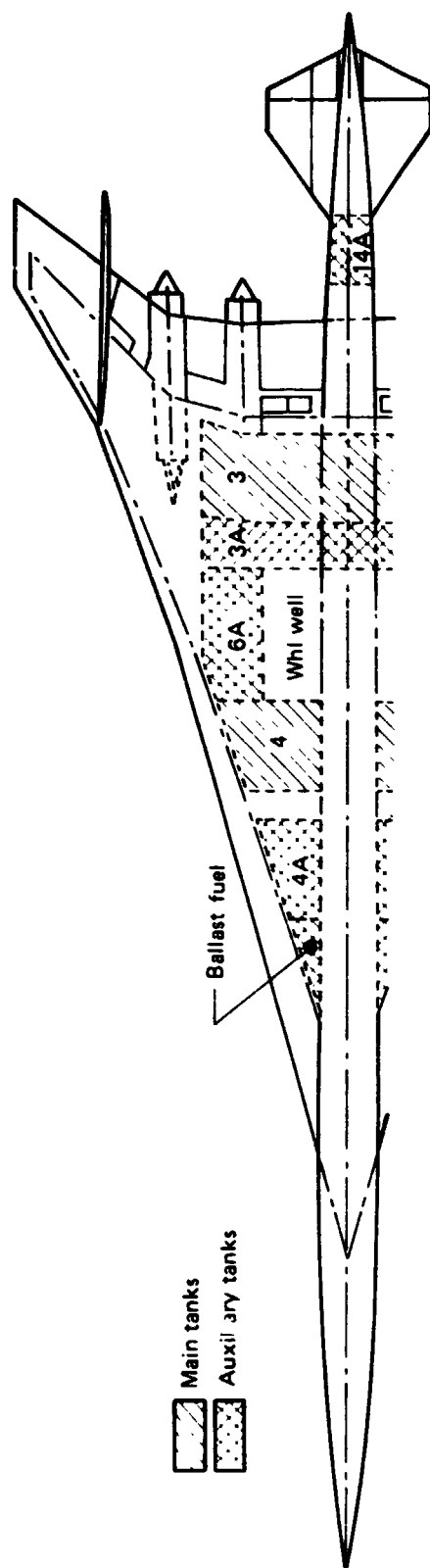


FIGURE 1-28.--FUEL TEMPERATURE VS TANK CONDUCTANCE



Tank no.		Capacity/airplane		cg body station		Average depth	
Left	Right	kg	lb	m	in.	m	in.
1	4	45,495	100,300	54.84	2,159	1.10	43.3
2	3	45,450	100,200	70.59	2,779	0.73	28.7
1A	4A	28,894	63,700	47.85	1,884	1.21	47.6
2A	3A	30,481	67,200	66.70	2,626	0.94	37.2
5A	6A	27,805	61,300	61.47	2,420	0.80	31.5
14A		15,059	33,200	84.05	3,309	2.65	104.5
Ballast		13,154	29,000	42.77	1,684	1.27	50.0
Total capacity (excluding ballast)		193,184	425,900	62.60	2,464.6		

FIGURE 1-29. — FUEL TANK LAYOUT—MODEL 969-512B

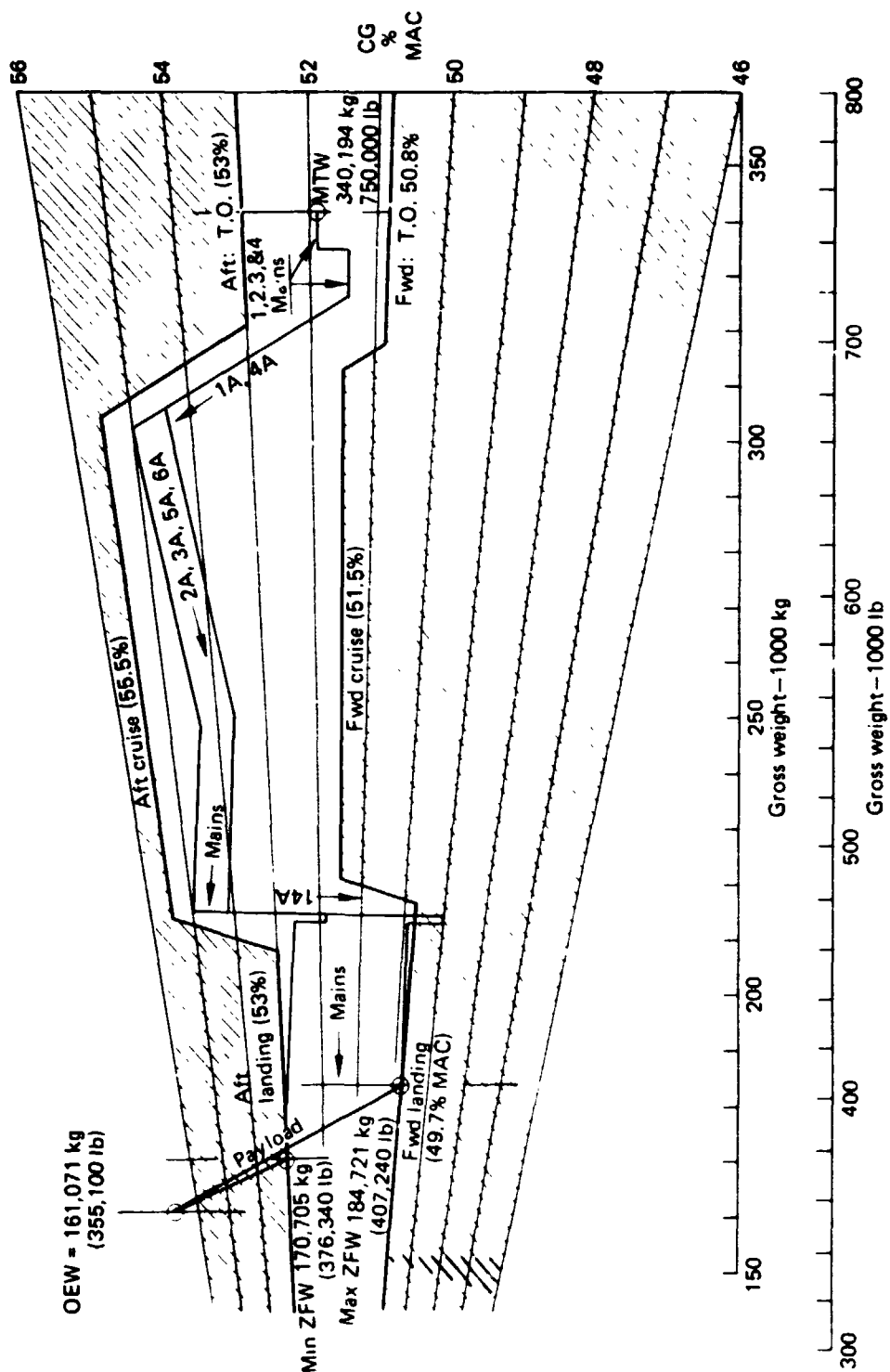


FIGURE 1-30. FUEL MANAGEMENT—MODEL 969-512B

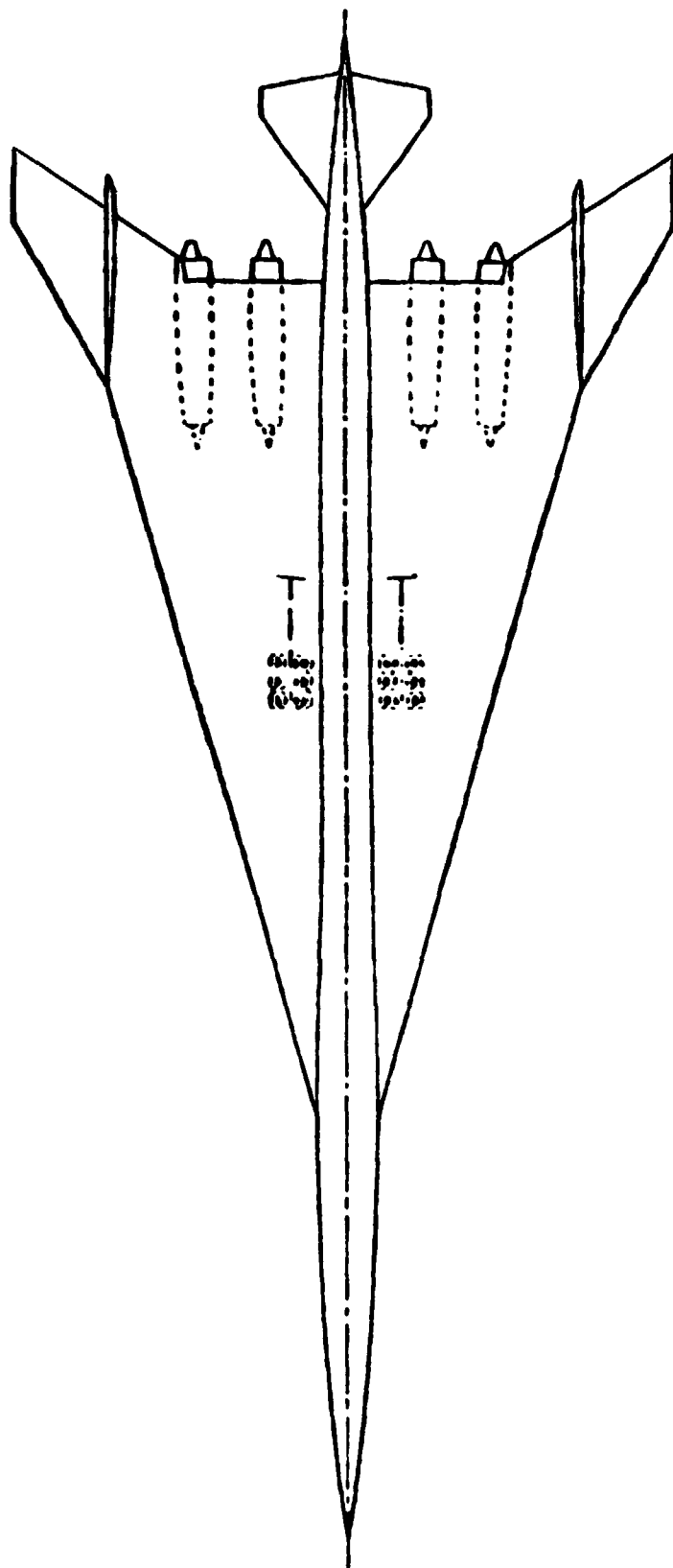


FIGURE 1.31. -- CONFIGURATION FOR ENGINE PLACEMENT STUDY, OUTBD ENGINE FWD

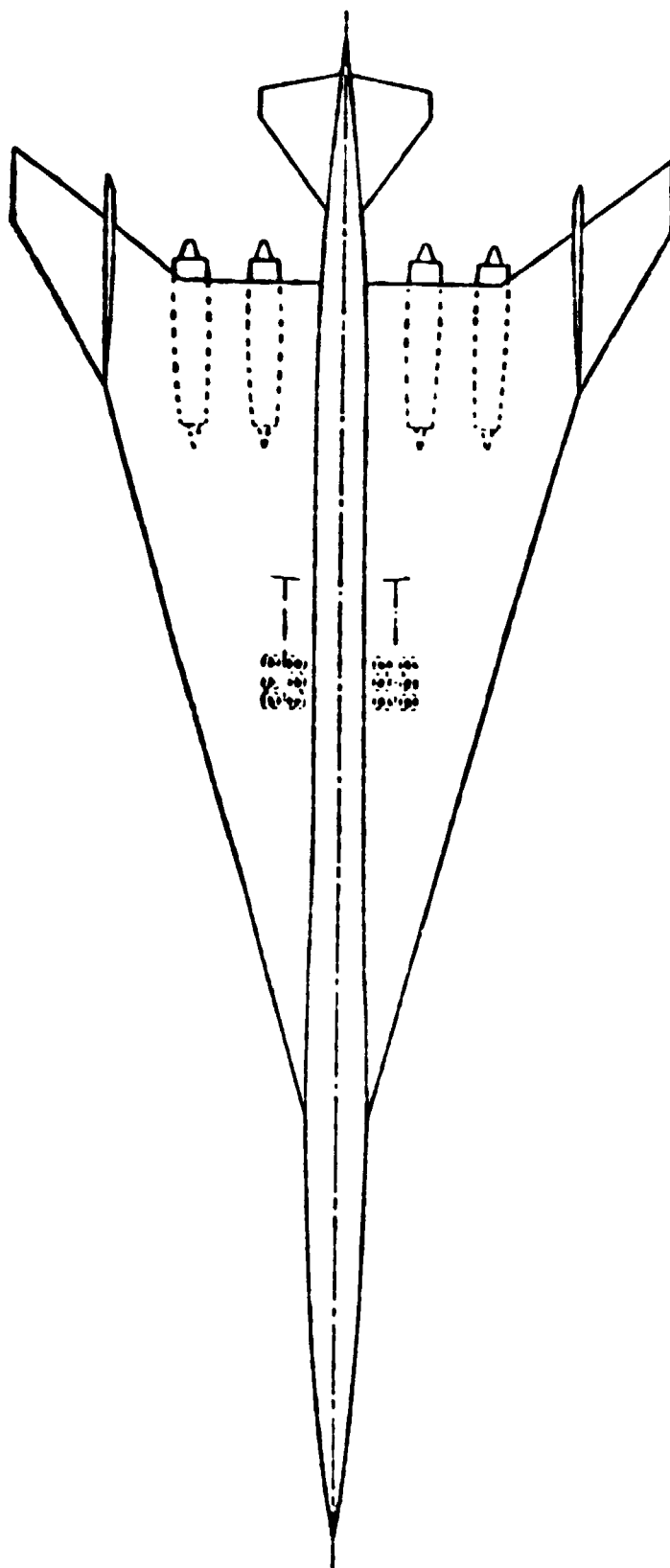


FIGURE 1-32 CONFIGURATION FOR ENGINE PLACEMENT STUDY, ENGINES INBD

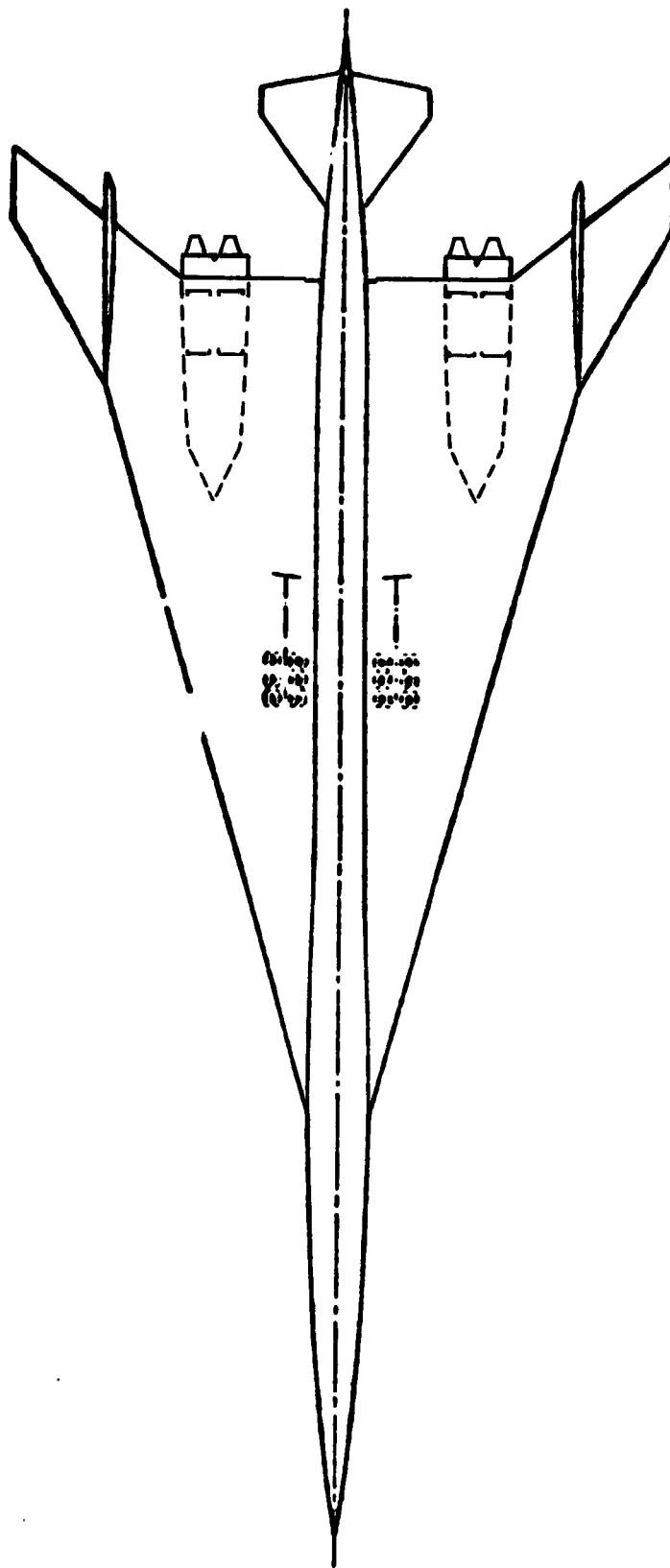


FIGURE 1-33. — CONFIGURATION FOR ENGINE PLACEMENT STUDY, DUAL PODS

SECTION 2

STRUCTURAL CONCEPTS DEFINITION

by

H. M. Tomlinson

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Symbols and Abbreviations

N_x	Stress resultant in the x direction
N_y	Stress resultant in the y direction
N_{xy}	Stress resultant shear
$\bar{t}$	Equivalent thickness

SECTION 2

STRUCTURAL CONCEPTS DEFINITION

The object of the Task I structural evaluation is to develop and screen candidate structural concepts and arrangements consistent with the requirements of a Mach 2.7 supersonic transport and a 1975 program go-ahead. The data resulting from the structural evaluation is the basis for making the specific structural concept recommendations for the 969-512B configuration shown in figure 1-20.

STRUCTURAL CONCEPTS STUDY GUIDELINES

The components of the airplane chosen for structural evaluation are those having the largest potential impact on the structural weight of the airplane; specifically, these are the wing skin panels, wing internal box structure, and the body shell. A baseline for comparison was established for each of these components from the previous study of the 969-336C airplane. Using these components from the 969-336C as a baseline, alternate structural concepts and arrangements were configured for each of these major components. The alternate structural concepts were analyzed for weight, manufacturing complexity, stiffness, fatigue, thermal conductance and material cost, and qualitatively evaluated for maintainability and fail-safe requirements. This evaluation provides the basis for making specific structural concept recommendations for the 969-512B configuration.

ESTABLISHMENT OF BASELINE

To establish a consistent data base for comparing each of the structural concepts a baseline set of design loads and environmental conditions was established from the 969-336C (SCAT-15F) study. Three locations on the wing, as shown in figure 2-1, and four locations along the body, as shown in figure 2-2, were chosen as control points. These points represent a typical range of structural requirements for an Arrow Wing configuration. The critical design conditions, loads, and thermal environment were established for each of the control locations for the wing and body. A typical set of body conditions is shown in figure 2-3.

BASELINE-WING SKIN PANELS

The baseline wing skin panels on the 969-336C at design point 249 and the lower surface at point 269 utilized titanium "stressskin honeycomb" with welded edge members as shown in figure 2-4. Stressskin is a spotwelded honeycomb configuration which has constant skin gages and no edge member when purchased from the supplier. The addition of welded edge members, milling of face sheets, and contouring of the wing skin panels was required to complete a typical skin panel assembly. A face sheet of .060 inch 6Al-4V titanium was the upper limit of manufacturing capability as evaluated by Boeing. Since the loads at point 431 and the upper surface at point 269 require face sheets greater than .060 inch thick, the corrugated core sandwich is used as the

baseline. Panel face sheets are machined to provide integral edge attachments. The panels are assembled by spot welding at the corrugation nodes. These panels are structurally efficient for loads oriented axially with the corrugation but are relatively inefficient for transverse loads.

BASELINE-WING SPARS

The baseline spar configuration utilizes riveted sheet stiffeners as shown in figure 2-5 at point 249. At points 269 and 431 welded sine wave spars, as shown in figure 2-6, were utilized. The welded sine wave spar is a very efficient configuration for both vertical shear and fuel crash loads when accessory attachments and system cutout reinforcements are minimized.

BASELINE-BODY PANELS

The 10 body panel control points utilize a semi-monocoque arrangement of riveted sheet stiffener construction. The sizing, as a baseline for comparison, is shown in figure 2-7. These structural control points were used as a basis for evaluation of the alternate concepts. Design conditions shown in figure 2-3, identical to those for the baseline, were then used to develop alternate design concepts.

ALTERNATE DESIGN CONCEPTS

Figure 2-8 shows the alternate structural concepts which were designed for the wing skin panels, wing spars, and body shell. Utilizing the material selections recommended from the material screening evaluation, and design loads and environmental conditions identical to the comparable baseline component, each candidate specimen was designed and sized to maintain a margin of safety from 0 to 5%. Static, thermal, and fail safe requirements were incorporated into the design of each specimen.

WING SKIN PANELS

Silver Brazed Steel Honeycomb.

Silver brazed steel honeycomb as shown in figure 2-9 was investigated because of its potential structural efficiency at high temperatures and high end loads. This configuration was sized for both points 269 and 431. It was not investigated at point 249 because the low loads produce a minimum gage sizing which is heavier than similar concepts of minimum gage titanium.

The configurations utilize PH15-7Mo steel face sheets with 1/4 inch cell-5.5 pcf honeycomb core also of PH15-7Mo steel. A dense core of 35 pcf-1/8 inch cell is used around the periphery of the panel to resist bolt crushing loads from panel installation fasteners. The dense core is attached to the field core by spot welding of core nodes. The outside face sheets are machined to provide a step and skin pad for the installation of the exterior panel splice plate. The fit up of the dense core to the face sheet contour is a critical manufacturing requirement. The complex fit up of parts and brazing tool requirements produce a complexity rating of 189 as shown in table 4-1 compared to the

corrugated core sandwich baseline complexity of 100. The steel brazed honeycomb has good biaxial capability for N_x and N_y loads but is still less efficient than brazed titanium honeycomb configurations because of its lower strength to weight ratio.

Aluminum Brazed Titanium Honeycomb.

The structural efficiency of aluminum brazed titanium honeycomb throughout the range of Arrow Wing end loads justifies its development as a concept candidate. The configuration as shown in figure 2-10 was sized for a three control point locations (249, 269 and 431). The configuration utilizes Ti-6Al-4V Condition I for the skin face sheets and 1/4 inch cell-4.9 pcf field core of Ti-3Al-2.5V. Dense core of approximately 14 pcf for 3/16 fasteners and 28 pcf core for 1/4 fasteners is used around the panel edge. The panel is similar in configuration to that described for the steel brazed arrangement. Fit up of the core to match the skin panel recess is again a critical manufacturing requirement. The brazing cycle is somewhat less complex for aluminum brazed titanium than for steel and accounts for the lower manufacturing complexity of 179 (shown in table 4-1). The aluminum brazed titanium core is structurally efficient in reacting combined N_x , N_y , and N_{xy} loads and is a superior candidate because of the efficiency of honeycomb in the load ranges involved and the high strength to weight ratio of the Ti-6Al-4V material.

Machined-Waffle Panel.

The waffle skin panel arrangement was chosen as a structural concept candidate because of improved biaxial capability compared to conventional sheet stiffener arrangements. The waffle panel arrangements as shown in figure 2-11 are fully end milled from Ti-6Al-4V plate. The material allowables are less than those for sheet stock because of the plate thickness. The panels are relatively simple to manufacture since only machining is required, however, extensive numerically controlled machine facilities would be required to support production airplane quantities. Two configurations were investigated with the grid oriented parallel to the N_x and N_y directions and with the grid on a 45° bias. The bias arrangement was found to provide a slightly superior weight efficiency at the higher end loads because the web stiffeners were carrying some of the shear and axial loads in truss action. The machined waffle configuration was rated at a manufacturing complexity of 125 compared to the baseline corrugated core complexity of 100. The waffle panel arrangements are of average structural efficiency for tension applications and below average efficiency for compression applications.

Riveted-Sheet Stiffener.

Riveted sheet stiffener construction was investigated for points 269 and 431 because of its manufacturing simplicity and relatively good tension efficiency. The sheet stiffener arrangement as shown on figure 2-12 utilizes machined zee sections riveted to a machined skin. Electromagnetic riveting of zeeps to skin is incorporated to achieve a high fatigue rating. The stiffeners and skin material are made of 6Al-4V titanium. This arrangement achieves a manufacturing rating of 54, the best of any concept investigated compared to the baseline complexity of 100. The weights analysis indicates

the sheet stiffener arrangement to be fairly competitive for tension applications, but it cannot compete with honeycomb arrangements in compression applications in the load ranges involved.

Machined and Welded-Sheet Stiffener.

The integrally machined and welded sheet stiffener arrangements as shown in figure 2-13 have been configured to improve the transverse buckling allowables. The configurations have been sized for points 269 and 431 and have utilized Ti-6Al-4V Condition I. For both arrangements shown, the skin panel plate is machined with elevated ridges and the machined stiffener angles are then butt welded to the outstanding ridges. The ridges are dimensioned to provide adequate clearance for the welding head to have direct access for a simple butt weld. The manufacturing complexity ratings are 84 and 94 for the two configurations compared to the baseline of 100. Those configurations in tension or with small N_y load do not utilize any chordwise stiffener. A chordwise stiffener is utilized at point 269 upper, however, to improve the transverse buckling allowable. The material which would normally be used in the stiffener skin flange has been redistributed into the sheet to improve the panel buckling allowable. The machined and welded sheet stiffener arrangement is a superior candidate for tension application. It is not as efficient as honeycomb for compression application in the load ranges involved at points 269 and 431.

WING SPARS

Machined and Welded-Sheet Stiffener.

The machined and welded sheet stiffener spar concept was initiated as a means of eliminating web to chord overlaps and stiffener-web overlaps, thereby improving the structural efficiency over that achieved in conventional riveted sheet stiffener construction. The configuration as shown in figure 2-14 was sized for points 249, 269, and 431 utilizing Ti-6Al-4V Condition I. The web is pocket chem-milled with integral pads for joining chord members and stiffeners. The basic web is milled to .015 gage. The machined web stiffeners, and spar chords are joined to the web by fusion welding. Electron beam welding at approximately 45° to the web is utilized for installation of the web stiffeners. The web and spar chord detail parts are trimmed to allow for weld shrinkage to comply with final contour requirements. Overlapping of spar chord and web, and stiffener and web as required in riveted construction, is eliminated. The web skin pads are utilized as part of the effective web stiffener material. Because of the elimination of joint overlaps and efficient use of material, this configuration is the most weight efficient of the sheet stiffener concepts.

Machined Extrusion-Sheet Stiffener.

The machined extrusion spar as shown in figure 2-15 was investigated as a means of approaching the structural efficiency of the machined and welded spar arrangement while minimizing cost by reducing welding requirements. The machined extruded spar arrangement was sized for points 249, 269 and 431 using 6Al-4V titanium. The titanium web stiffener extrusion requires 100% machining because of surface

contamination and extrusion gage limitations. The maximum span per extruded web segment is approximately 21 inches. Fusion butt welding would be utilized to join web segments. The integrally machined web and stiffener eliminates any other requirement for welding. The web-spar overlap for riveting provides a fit up tolerance payoff for manufacturing, but also reduces the weight efficiency. The rivet overlaps place the weight efficiency of the machined extrusion web between the riveted and the welded sheet stiffener configuration. Because of the extensive machining required of the web stiffener combination it is assigned a slightly greater manufacturing complexity rating than the welded arrangement.

Aluminum Brazed Titanium Honeycomb.

The structural efficiency of both double edge and thin edge aluminum brazed titanium honeycomb configurations was investigated for spar application. Both configurations shown in figure 2-16 utilized SS2-30 Core (14.0) pcf at the periphery for bolt attachments. Brazing complexity has been minimized by placing all skin pads on the exterior eliminating any core fit up requirements. The thin edge configurations has used one welded chord attachment for minimum weight and one riveted for tolerance payoff. The double edge panel uses torque limited fasteners for spar chord installation. For both configurations it has been found that the two face sheets, core and braze alloy, required to carry the web shear, weigh more than an efficient single web such as sine wave configuration. The complexity of the brazing process, even for the simpler application, is considerably above the sheet stiffener or sine wave arrangements as shown in table 4-2.

Silver Brazed Steel Honeycomb.

The design and construction features shown in figure 2-17 are similar to those described for the aluminum brazed titanium honeycomb except for material selection. The skin and honeycomb core are of PH15-7Mo material. The structural efficiency is even less competitive than the titanium honeycomb construction because of its lower strength to weight ratio. The manufacturing complexity rating is higher for steel construction than titanium because the structural concepts were similar but the brazing cycle is more complex for steel.

BODY PANEL CONCEPTS

Each of the body structural concepts investigated was sized for the ten control points (four body sections) as shown in figure 2-2.

Aluminum Brazed Titanium Honeycomb.

The use of aluminum brazed titanium honeycomb skin panels for the body was investigated for the following reasons:

- Efficiency in carrying combined shear and compression is a requirement throughout most of the body end load range

- A large portion of the body structure is critical for compression load conditions.
- The high degree of inherent stability of honeycomb skins allows a reduction in support structure to meet general instability requirements.

The aluminum brazed titanium body skin design as shown in figure 2-18 utilizes a similar set of fabrication techniques to those required for the brazed wing skins. In addition, the panels must be brazed to curvature radii in the range of 68 to 455 inches with compound contour. The panels utilize integrally machined 6Al-4V titanium face sheets with SC4-20 commercially pure titanium field core. Dense core is used along the panel edge to resist bolt crushing loads. Edge core densities range from 14 pcf to 28 pcf depending on the panel fastener size. The outside skins are recessed for flush installation of circumferential splice straps only. Longitudinal splice straps are installed external to contour. The frame to panel attachments have been made by bolts which require densified core and skin pads locally at every frame. This requirement greatly complicates the panel fit up of core to face sheet. The added shell stability of the honeycomb panels has allowed use of 35 inch frame spacing. The complexities of the fuselage panel fit up and the brazing process yield a manufacturing complexity rating of 365, as shown in table 4-3, compared to the baseline sheet stiffener complexity of 100.

Integrally Machined-Sheet Stiffener.

Integrally machined body skin panels have been investigated as a means of increasing structural efficiency by more efficient tailoring of the cover material to the bending and shear requirements of the body. Stiffener spacing and intervening nodes as shown in figure 2-19 have been sized and spaced to optimize compression and shear allowables. Integrally machined nodes for stiffener attachment eliminate the requirement for a riveted stiffener skin flange. Machined angle stiffeners have been welded to the face sheet node by fusion butt welding. Integral circumferential skin pads are utilized as part of body frame outer chord. A separate frame fail safe chord is located inside of the skin stiffeners. All skin and frame components have been designed of 6Al-4V Condition I titanium. The assembly of the monolithic panels by machining and welding represents a slightly more complex manufacturing procedure than the machining and riveting sequence utilized in sheet stiffener fabrication. This is indicated by the relative complexity factor of 158 as shown in table 4-3.

Corrugated Core Sandwich.

An investigation of corrugated core sandwich body covers has been made as an alternate cover arrangement. Corrugated core sandwich as shown in figure 2-20 is of interest for body application because the major bending loads are axial with the core direction, and the skins as well as core both act as effective bending material. The arrangement was designed of 6Al-4V Condition I titanium for both core and face sheets. The core configuration of constant 1.27 inch pitch and .75 inch height is electron beam welded to the face sheets at the core nodes. Longitudinal tear straps are located inside the inner skin and between the core trusses. Chem-milled skin pads are located at the frames and serve as part of the body frame chord material. The outer frame chord is electron beam welded integrally to the skin pad with no discontinuities except at panel splices.

The double compound contour required for the body panels greatly complicates the core forming and panel assembly tasks. Since the core pitch is constant and the skin panels are tapered in planform, individual core segments run out along the side of the panels with complicated splices required. These complexities are all reflected in the high manufacturing producibility factor of 380 given the structural arrangement. Based on the high manufacturing producibility factor and structural splice complexities no further work was completed on this concept.

ALTERNATE STRUCTURAL ARRANGEMENTS

WING STRUCTURAL ARRANGEMENT-SHEET STIFFENER PANELS

Shown in figure 2-21 is the general structural arrangement of the inspar wing for installation of sheet stiffener skin panels. The multi-rib sheet stiffener installation was investigated only for the more highly loaded portion of the wing. A multi-rib structural arrangement at approximately 28 inch rib spacing has been utilized. Skin panels are attached to the ribs by bolts through the inner stiffener flanges. Ribs have been positioned to back up engine support beams and wheelwell ribs.

WING SPAR SPACING OPTIMIZATION STUDY

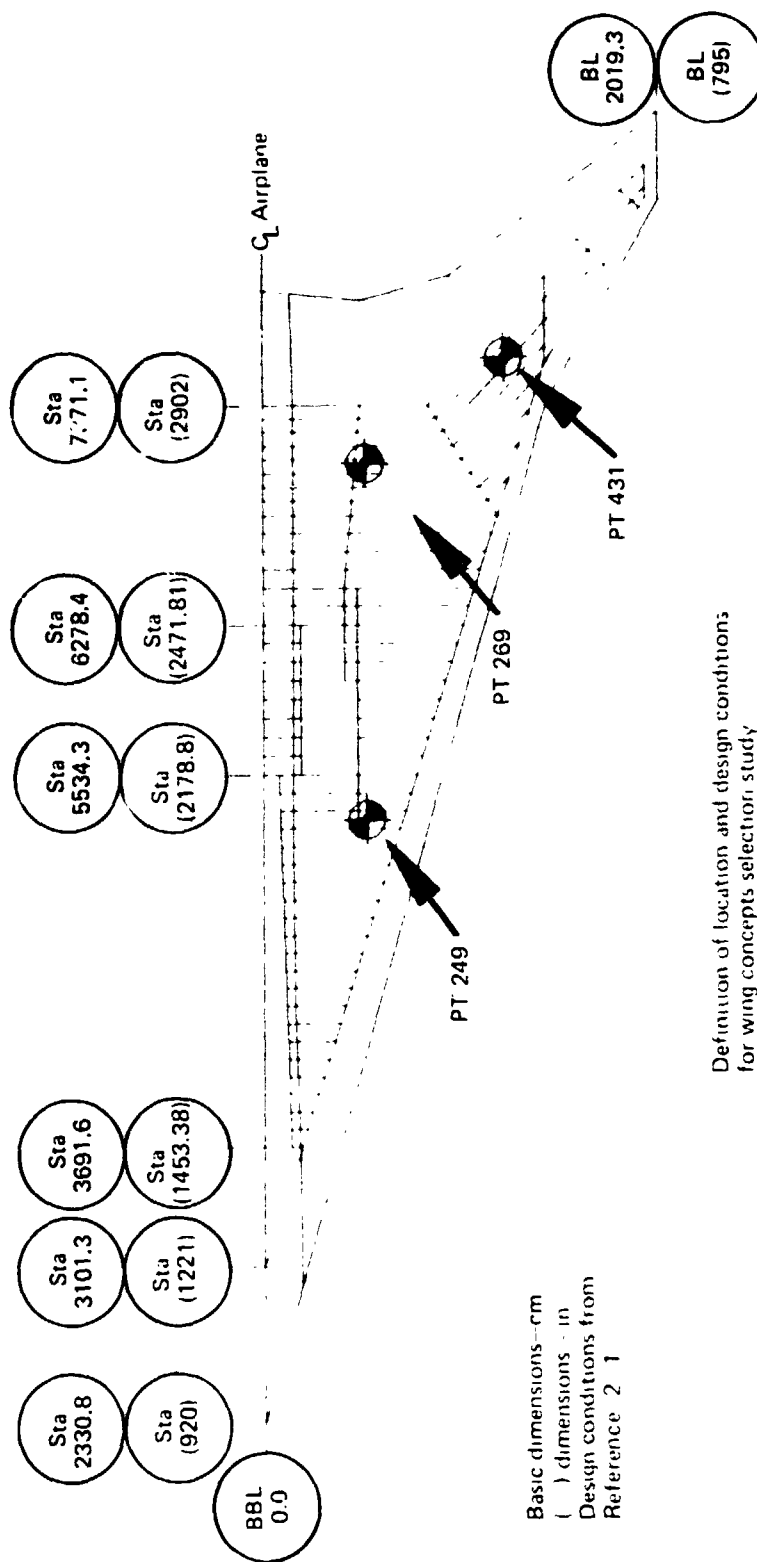
Analysis of the minimum weight multi-spar spacing has been investigated for the compression panel design conditions at points 269 and 431. An equivalent $\bar{t}$, including wing skins, edge members, and spar has been sized for the combined design loads at these points. Figure 2-22 shows t^1 at the minimum weight spar spacing is approximately 35 inches for both design points. This data was used as a basis for selecting a 35 inch spar spacing for the structural arrangement of the 969-512B.

WING STRUCTURAL ARRANGEMENT-POST AND SPAR

Post and spar arrangements have been investigated for spar spacings of both 20 and 40 inches. Each arrangement utilized a row of posts midway between the spars. The smaller spar spacing tends to provide the more efficient structural arrangements but internally is space limited. A 40 inch spar spacing as shown in figure 2-23 was investigated. Preliminary study indicated a possible weight saving of 5% of installed panel weight; however, the space restriction of the intervening post was found unacceptable. Wing assembly, maintenance of fuel sealing, and insulation systems was found to be inadequate due to space limitations of the posts.

REFERENCE

- 2-1 *Mach 2.7 Fixed Wing SST Model 969-336C (SCAR-15F)*. Boeing document D6A11666, July 11, 1969.



Point	Component	Design condition	T C	T (F)	N _x kN/m	N _x (kips/m.)	N _y kN/m	N _y (kips/m.)	N _{xy} kN	N _{xy} (kips/m.)
249	Upr panel	Refuel overpress	Rt	Rt						
249	Lwr panel	Refuel overpress	Rt	Rt						
269	Upr panel	Trans descent (1.2-3)	121° 15'	250° 15'	-1908	-10.9	-259	-1.48	1105	0.32
269	Lwr panel	Trans climb (1.2-1)	Rt	Rt	2070	11.82	357	2.04	1205	6.89
431	Upr panel	Trans climb (1.2-1)	Rt	Rt	-3820	-21.8	-184	-1.05	290	1.66
431	Lwr panel	Trans climb (1.2-1)	Rt	Rt	3820	21.8	184	1.05	290	1.66

FIGURE 2-1. - WING - DESIGN CONDITIONS AND CONTROL LOCATIONS

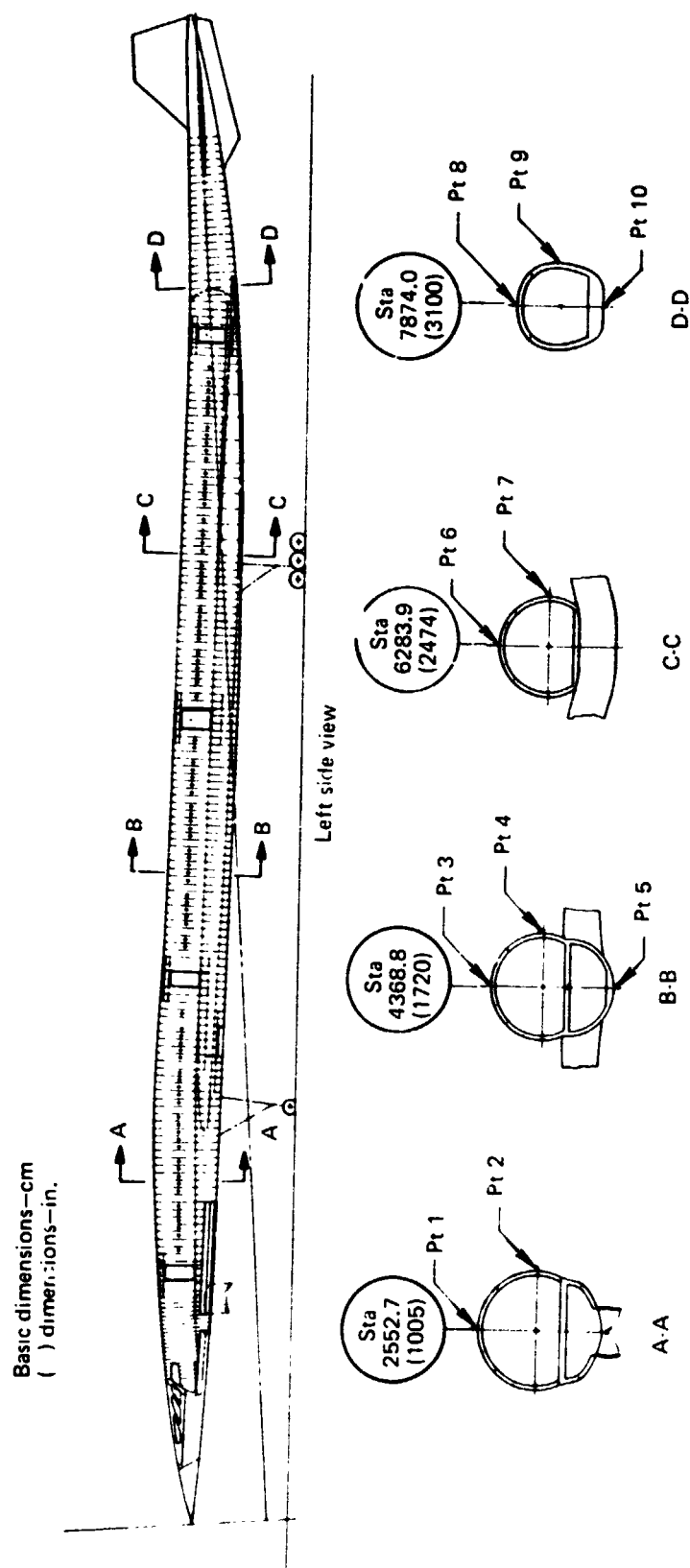


FIGURE 2-2.—BODY—CONTROL LOCATIONS

Load	Design condition					Load			P-Δ geometry			Panel		
	Dist. p	Max. m	Eq. v.	Int. m	Load factor	Temp.	Bending moment	Shear	Rad.	Depth	Shear area (T _v)	Shear	T (T _v)	Load
1	Beam size													
	2,552 x 10													
2	Beam size													
	1,045,000													
3	Beam size													
4	Beam size													
5	Beam size													
6	Beam size													
7	Beam size													
8	Beam size													
9	Beam size													
10	Beam size													

Use appropriate No. units.
Temp. in °C, loads in mm.

FIGURE 2-3.—BODY—DESIGN CONDITIONS

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Spotweld
Electron beam weld
83.29 kg/m³ (5.2 pcf) Ti-3Al-2.5V
Ti-6Al-4V Cond I

FIGURE 2.4.—WING-BASELINE SKIN PANELS

See Figure 2-1 for panel locations
Basic dimensions—cm
() dimensions—in.

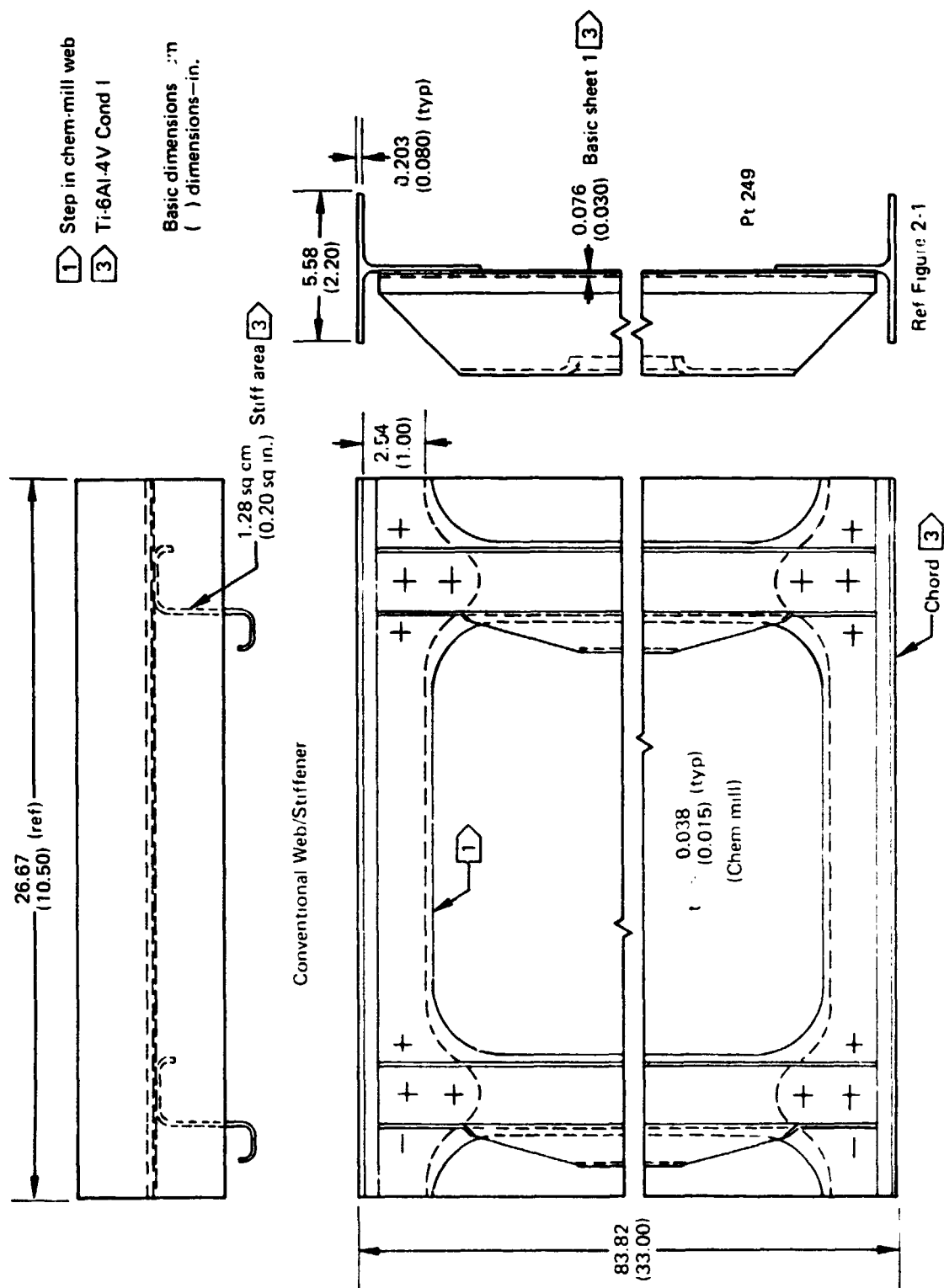


FIGURE 2-5.—WING SPAR—BASELINE (PT 249)

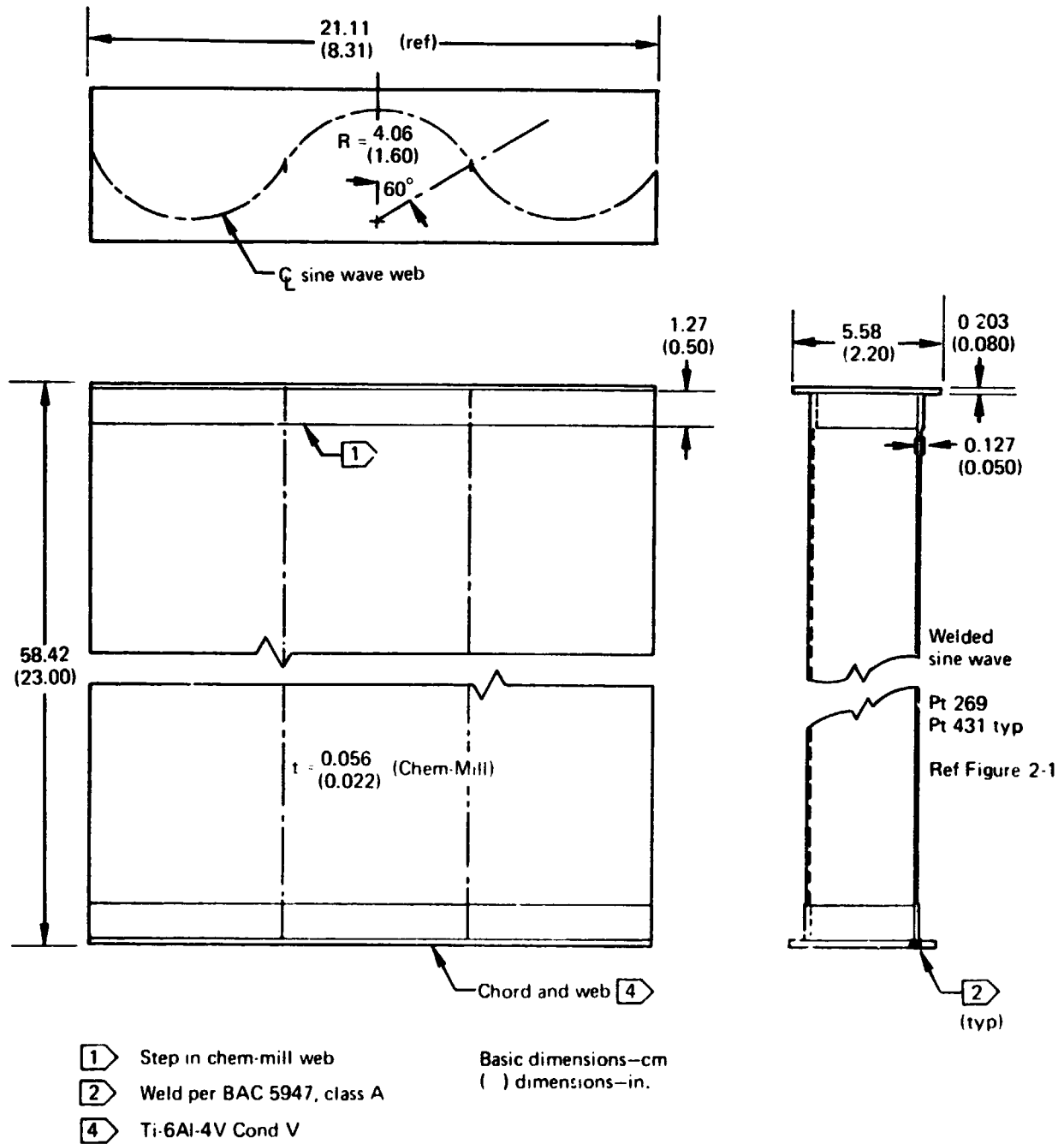
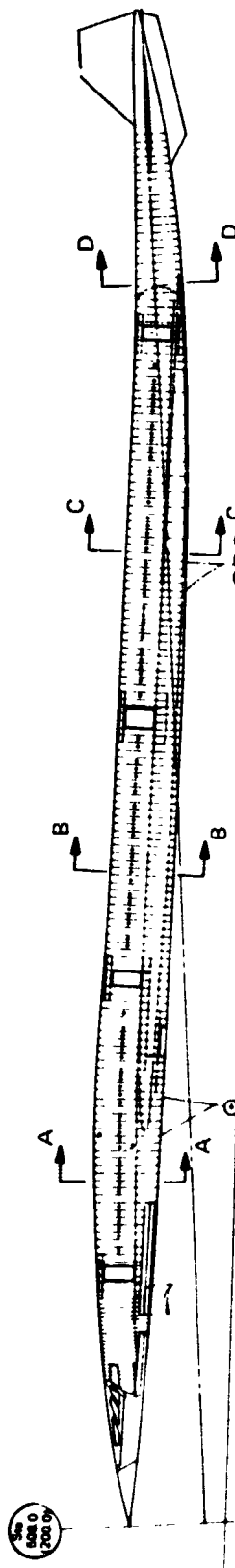
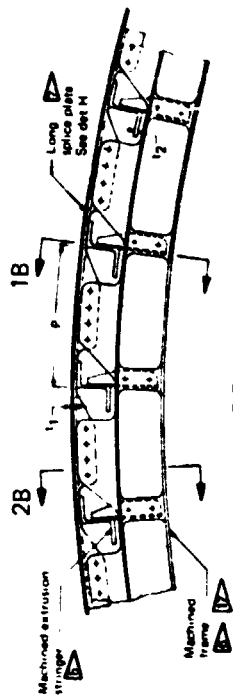
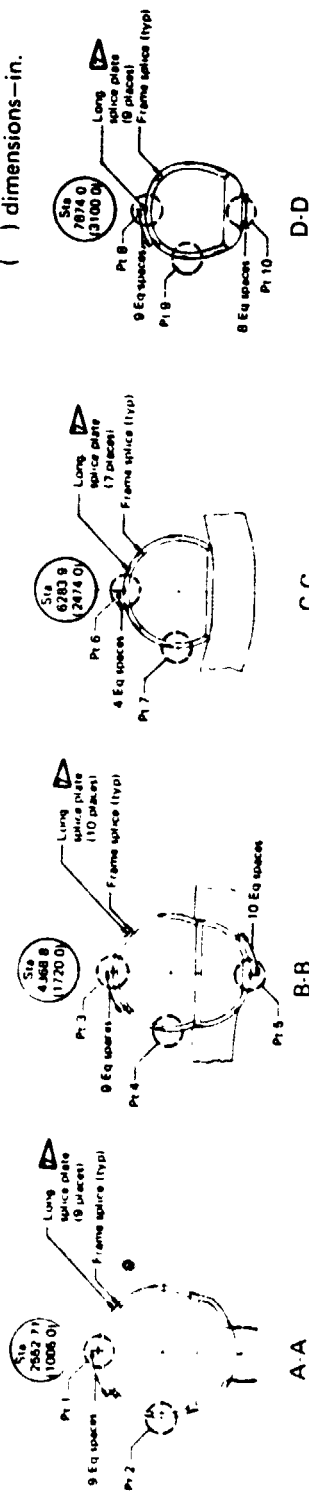


FIGURE 2-6.—WING SPAR—BASELINE (PT 269 AND 431)

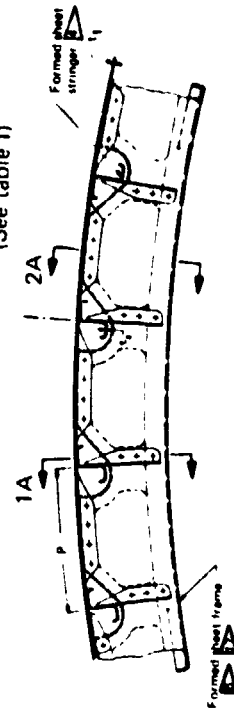


Basic dimensions—cm
() dimensions—in.

LEFT SIDE VIEW



DETAIL B
(See table I)



DETAIL A
(See table I)

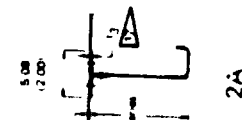
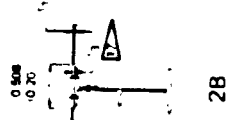
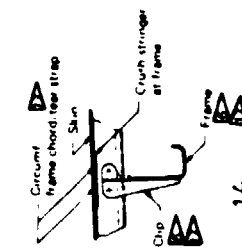
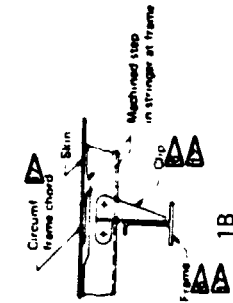
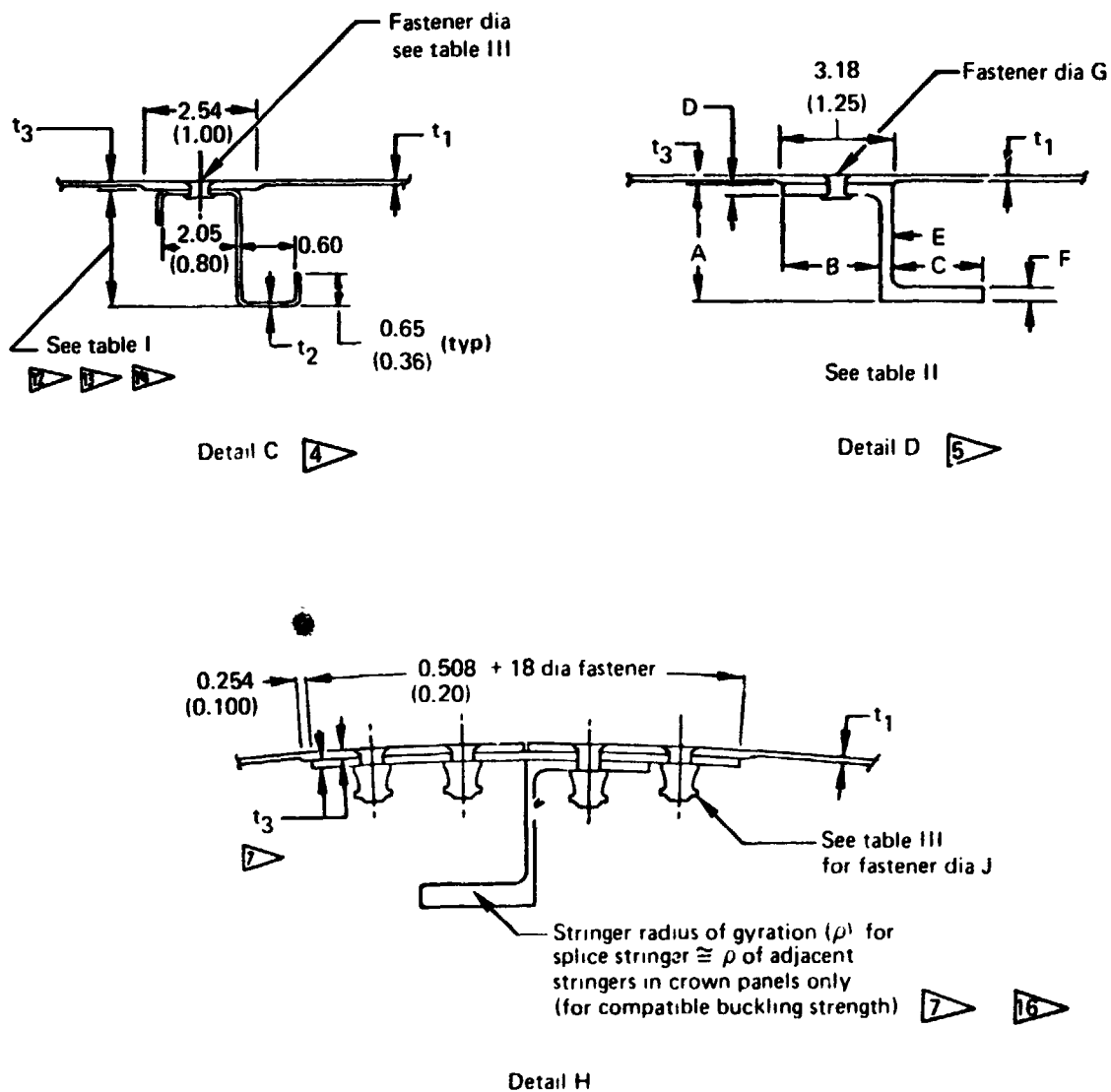


FIGURE 2-7—BODY—BASELINE SHELL ARRANGEMENT

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Basic dimension - cm
() dimension - in.

Figure 2-7.-BODY-BASELINE SHELL ARRANGEMENT (CONTINUED)

Pl no.	Body sta cm (in.)	Detail Frame	Skin t ₁	Stringer t ₂	Stringer area	stringer spacing	Effective area pad up and splice mtl	Total effective	Panel load x 10 ³ N/M (lb/in)	Design load condition	Panel weight
	Circumf location		Stringer	t ₃ cm (in.)	t ₂ cm (in.)	cm ² (in.)	cm (in.)	cm ² (in ²)	t cm (in.)		
	Sta (2,562.70) 1,005.00				11		7 8	7 17			
1	Upr crown 2	A	0.076 (0.030)	0.076 (0.030)	0.761 (0.118)	13.97 (5.50)	2.41 (0.374)	0.142 (0.056)	435 (2,485)	ten	Low angle of attack Max canard Dynamic landing lg cruise panel stab
		C	0.137 (0.054)	0.137 (0.054)	12				348 (1,988)	comp	
2	Side panel 3	A	0.076 (0.030)	0.076 (0.030)	0.761 (0.118)	12.70 (5.00)		0.136 (0.054)	106 (605)	shear	
		C	0.137 (0.054)	0.137 (0.054)	13			9	23 (133)	shear	
	Sta 4,368.80 (1,720.00)										
3	Upr crown 2	A	0.1016 (0.040)	0.152 (0.060)	1.393 (0.216)	12.70 (5.00)	2.99 (0.464)	0.224 (0.088)	1,164 (6,646)	ten	End of cruise Max canard End of cruise lg cruise panel stab
		C	0.137 (0.054)	0.137 (0.054)	13				957 (5,570)	comp	
4	Side panel 3	A	0.076 (0.030)	0.076 (0.030)	0.761 (0.118)	12.70 (5.00)		0.136 (0.054)	155 (888)	shear	End of cruise lg cruise panel stab
		C	0.137 (0.054)	0.137 (0.054)	12			9	46 (261)	shear	
5	Lwr crown 3	A	0.127 (0.050)	See table II A	3.32 (0.515)	11.18 (4.40)	6.58 (1.010)	0.456 (0.178)	2,087 (11,921)	comp	End of cruise lg cruise panel stab
		D	0.183 (0.072)	See table II A	16				613 (3,500)	shear	
	Sta 6,283.90 (2,474.00)										
6	Upr crown 2	B	0.203 (0.080)	See table II A	3.258 (0.505)	11.13 (4.50)	6.76 (1.048)	1.432 (0.222)	4,827 (27,564)	ten	3g taxi Negative maneuver
		D	0.254 (0.100)	See table II A	16				1,532 (8,750)	comp	
7	Side panel 3	B	0.228 (0.090)	See table II A	1.529 (0.231)	11.00 (4.50)		0.323 (0.127)	697 (3,980)	shear	3g taxi lg cruise panel stab
		D	0.228 (0.080)	See table II A	15			9	256 (1,460)	shear	
	Sta 7,874.00 (3,100.00)										
8	Upr crown 2	A	0.152 (0.060)	See table II A	2.19 (0.340)	11.18 (4.40)	5.13 (0.795)	0.961 (0.149)	2,376 (13,570)	ten	Transonic climb Negative maneuver Transonic climb lg cruise panel stab
		D	0.183 (0.072)	See table II A	16				1,954 (11,160)	comp	
9	Side panel 3	A	0.132 (0.052)	1.27 (0.050)	1.09 (0.169)	12.70 (5.00)		0.554 (0.086)	343 (1,960)	shear	lg cruise panel stab
		C	0.183 (0.072)	1.27 (0.050)	15			9	101 (575)	shear	
10	Lwr crown 3	A	0.2438 (0.096)	See table II A	0.983 (0.154)	11.68 (4.60)	6.051 (0.938)	0.495 (0.195)	2,737 (15,627)	comp	Transonic climb lg cruise panel stab
		D	0.2438 (0.096)	See table II A	16				802 (4,580)	comp	

Basic dimensions—cm
() dimensions—in.

FIGURE 2-7.—BODY-BASELINE SHELL ARRANGEMENT (Continued)

		Basic dimensions—cm () dimensions—in.								
Pt		A	B	C	D	E	F	G	t ₃	Area cm ² (in ²)
3	Splice stringer	3.18 (1.25)	2.03 (0.80)	1.52 (0.60)	0.305 (0.12)	0.203 (0.08)	0.254 (0.10)	0.397 (0.156)	0.137 (0.054)	1.652 (0.256)
5	Stringer	3.18 (1.25)	2.74 (1.08)	2.54 (1.00)	0.381 (0.15)	0.152 (0.06)	0.635 (0.25)	0.476 (0.188)	0.183 (0.072)	3.142 (0.487)
	Splice stringer	3.18 (1.25)	2.74 (1.08)	3.02 (1.19)	0.381 (0.15)	0.305 (0.12)	0.762 (0.30)	0.476 (0.188)	0.183 (0.072)	4.316 (0.669)
6	Stringer	3.18 (1.25)	2.74 (1.08)	2.54 (1.00)	0.305 (0.12)	0.305 (0.12)	0.508 (0.20)	0.635 (0.25)	0.254 (0.100)	3.097 (0.480)
	Splice stringer	3.18 (1.25)	2.74 (1.08)	2.54 (1.00)	0.305 (0.12)	0.305 (0.12)	0.508 (0.20)	0.635 (0.25)	0.254 (0.100)	3.097 (0.480)
7	Stringer	2.54 (1.00)	2.74 (1.08)	1.90 (0.75)	0.229 (0.09)	0.152 (0.06)	0.228 (0.09)	0.635 (0.25)	0.228 (0.090)	1.452 (0.225)
8	Stringer	3.18 (1.25)	2.74 (1.08)	2.54 (1.00)	0.305 (0.12)	0.152 (0.06)	0.305 (0.12)	0.476 (0.188)	0.183 (0.072)	2.097 (0.325)
	Splice stringer	3.18 (1.25)	2.74 (1.08)	2.54 (1.00)	0.279 (0.11)	0.254 (0.10)	0.535 (0.25)	0.476 (0.188)	0.183 (0.072)	3.187 (0.494)
10		3.18 (1.25)	2.74 (1.08)	2.54 (1.00)	0.381 (0.15)	0.152 (0.06)	0.381 (0.15)	0.635 (0.25)	0.244 (0.096)	2.497 (0.387)
	Splice stringer	3.18 (1.25)	2.74 (1.08)	2.54 (1.00)	0.381 (0.15)	0.152 (0.06)	0.635 (0.25)	0.635 (0.25)	0.244 (0.096)	3.142 (0.487)

Pt no.	Fastener dia cm (in.)
1	0.397 (0.156)
2	0.397 (0.156)
3	0.397 (0.156)
4	0.397 (0.156)
5	0.476 (0.188)
6	0.635 (0.250)
7	0.635 (0.250)
8	0.476 (0.188)
9	0.476 (0.188)
10	0.635 (0.250)

- 17 Circumferential body frame and skin pad up and body section joints not included in effective $\bar{t}$
- 16 See table for splice stringer area and geometry
- 15 This stringer is also used at the panel splice
- 14 Splice stringer depth is 3.18 (1.25) and t_2 is 0.1016 (0.04)
- 13 Stringer depth is 3.18 (1.25)
- 12 Stringer depth is 2.54 (1.00)
- 11 Local skin pad up is included in stringer area
- 10 Stringers not required
- 9 Pad up and splice matl on side panels is not included in total effective $\bar{t}$
- 8 Effective area shown is for one complete splice, includes skin pad up, splice plate and stringer
- 7 Longitudinal splice and pad up matl in crown panels is included in total effective $\bar{t}$
- 6 Frame and clip matl is Ti-6Al-4V Cond I
- 5 Extruded stringer matl is Ti-6Al-4V Cond I, I
- 4 Formed stringer matl is Ti-6Al-4V Cond I
- 3 Skin matl is Ti-6Al-4V Cond I
- 2 Skin matl is Ti-6Al-4V Cond I, I
- 1 Panel $\bar{t}$ shown is in Titanium

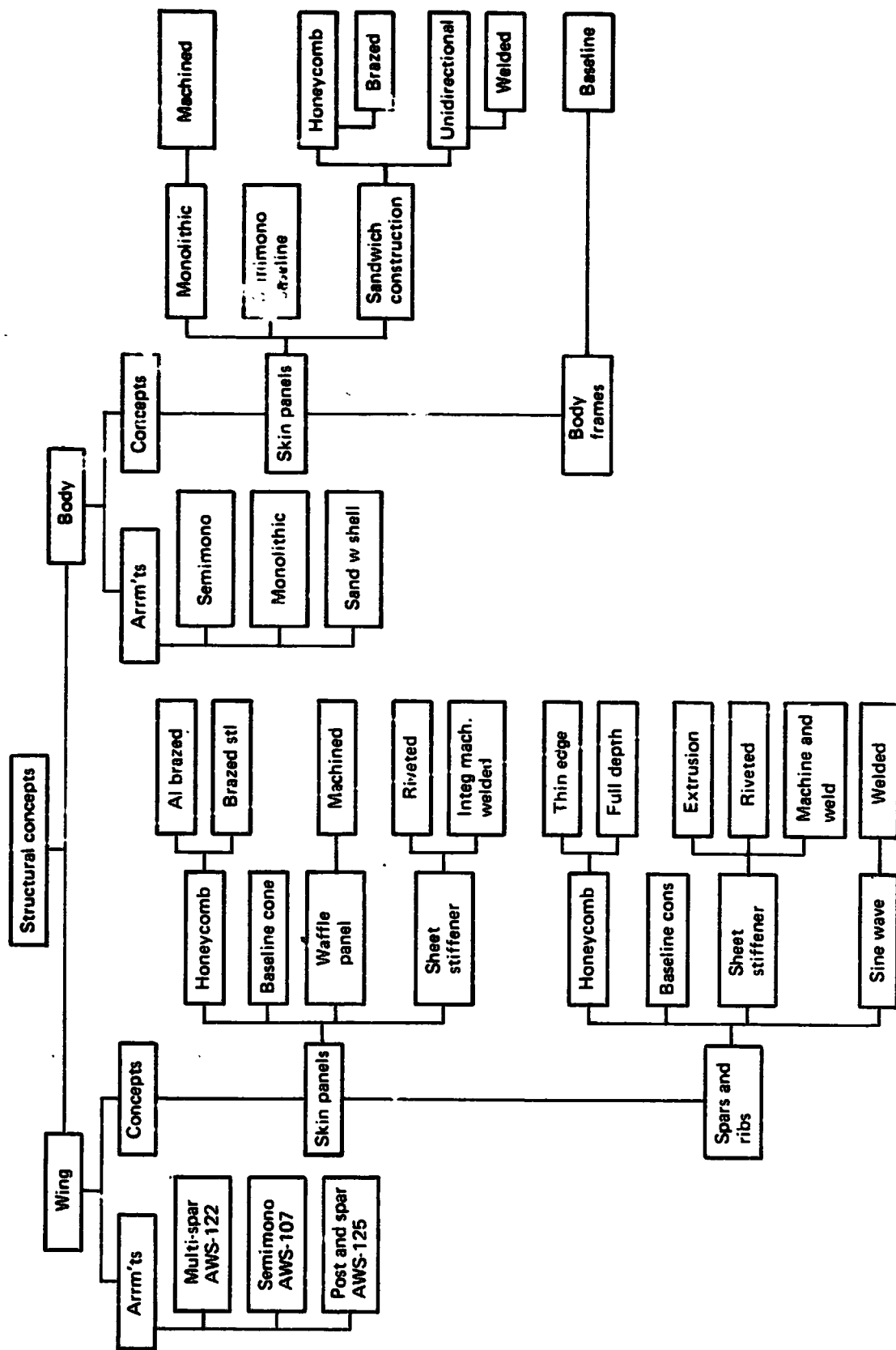
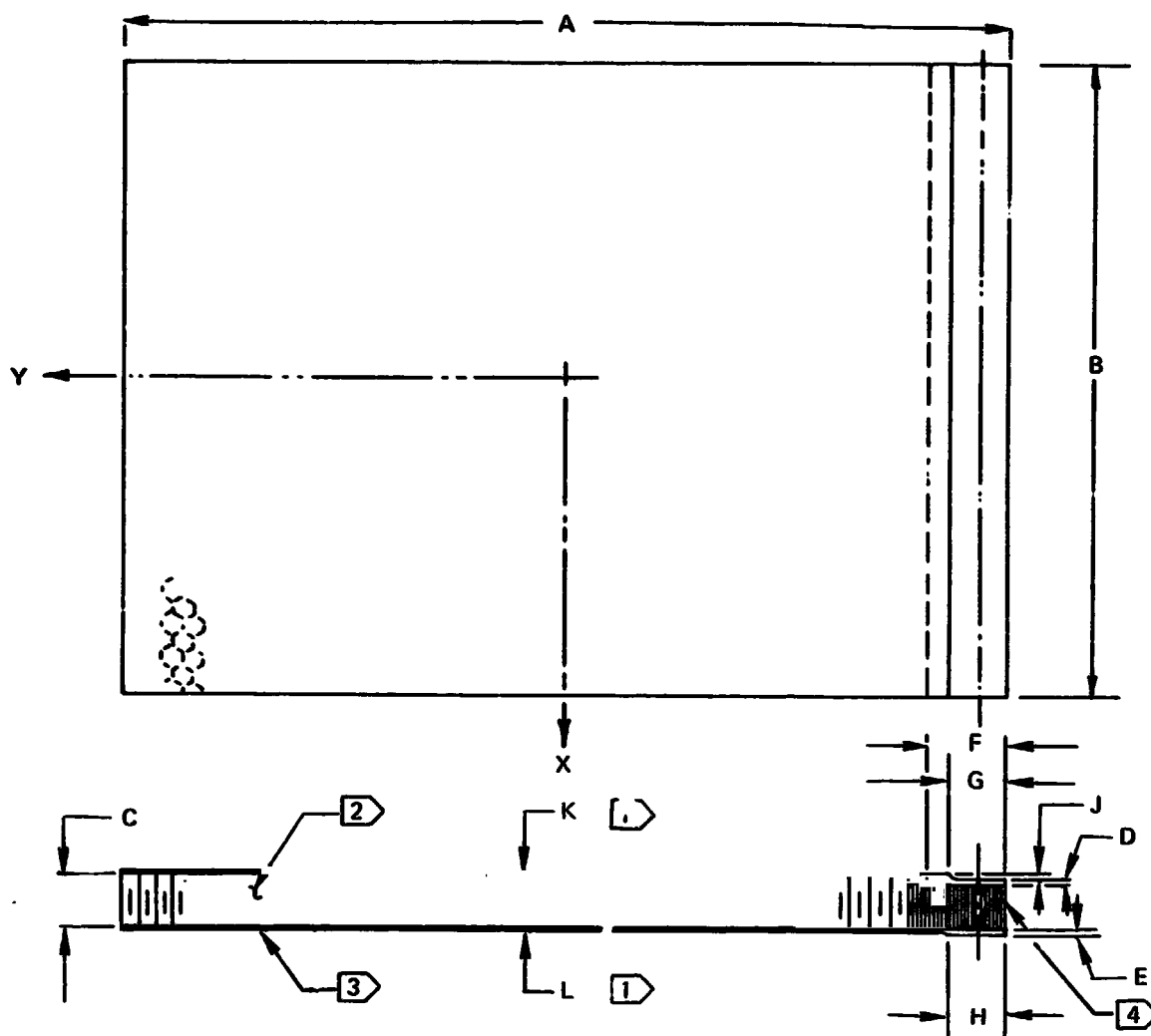


FIGURE 2-8. —ALTERNATE STRUCTURAL CONCEPTS

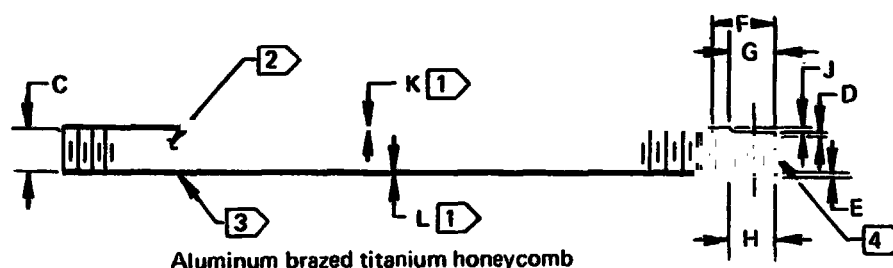
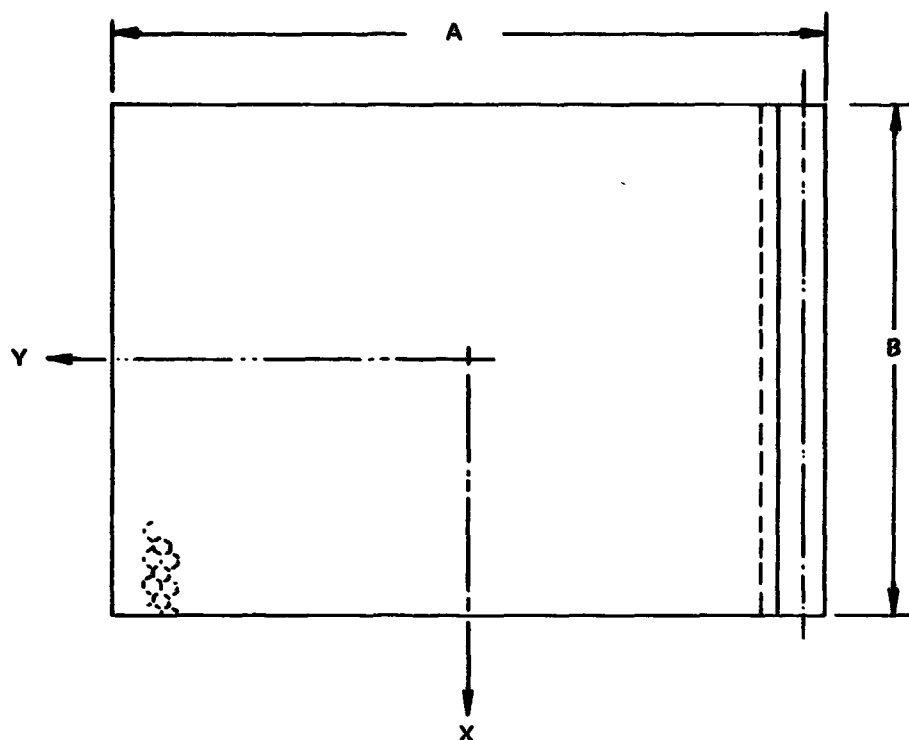


Point	A	B	C	D	E	F	G	H	J	K	L	
249												Upr Panel
												Lwr Panel
269	34.00 (13.38)	30.50 (12.00)	3.429 (1.35)	0.203 (0.080)	0.203 (0.080)	4.32 (1.70)	3.05 (1.20)	3.05 (1.20)	0.305 (0.120)	0.188 0.074	0.188 0.074	Upr Panel
										0.130 (0.051)	0.130 (0.051)	Lwr Panel
431	39.03 (15.37)	30.50 (12.00)	3.429 (1.35)	0.203 (0.080)	0.203 (0.080)	4.32 (1.70)	3.05 (1.20)	3.05 (1.20)	0.305 (0.120)	0.170 (0.067)	0.170 (0.067)	Upr Panel
										0.165 (0.064)	0.165 (0.064)	Lwr Panel

- 4 1/8 cell—561kg/m³ (35 pcf) PH 15-7 Mo
 3 Braze steel
 2 1/4 cell—88kg/m³ (5.5 pcf) PH 15-7 Mo
 1 PH 15-7 Mo TH 1030 $L_{tc} = 1.5$ mil

See Figure 2-1 for panel point locations
 Basic dimensions—cm
 () dimensions—in.

FIGURE 2-9.—WING SKIN PANEL—BRAZED STEEL HONEYCOMB



See Figure 2-1 for panel point locations

Basic dimensions—cm

() dimensions—in.

- 4 464.9 kg/m³ (29 pcf) SS2-60 core Ti-6Al-4V
- 3 Braze
- 2 78.50 kg/m³ (4.9 pcf)—SCA-20 core Ti-3Al-2.5V
- 1 Ti-6Al-4V Cond I

Point	A	B	C	D	E	F	G	H	J	K	L	
249	44.0	30.5	2.54	0.153	0.153	3.80	2.62	3.05	0.228	0.038 (0.015)	0.025 (0.010)	Upr Surf
	(17.32)	(12.00)	(1.00)	(0.060)	(0.060)	(1.50)	(1.03)	(1.20)	(0.090)	0.051 (0.020)	0.025 (0.010)	Lwr Surf
269	34.00	30.50	2.54	0.221	0.221	4.32	3.05	3.05	0.305	0.221 (0.087)	0.221 (0.087)	Upr Surf
	(13.38)	(12.00)	(1.00)	(0.087)	(0.087)	(1.70)	(1.20)	(1.20)	(0.120)	0.152 (0.060)	0.152 (0.060)	Lwr Surf
431	39.03	30.50	2.54	0.246	0.246	4.32	3.05	3.05	0.305	0.246 (0.097)	0.246 (0.097)	Upr Surf
	(15.37)	(12.00)	(1.00)	(0.097)	(0.097)	(1.70)	(1.20)	(1.20)	(0.120)	0.224 (0.088)	0.224 (0.088)	Lwr Surf

FIGURE 2-10.—WING SKIN PANEL—AL BRAZED TITANIUM HONEYCOMB

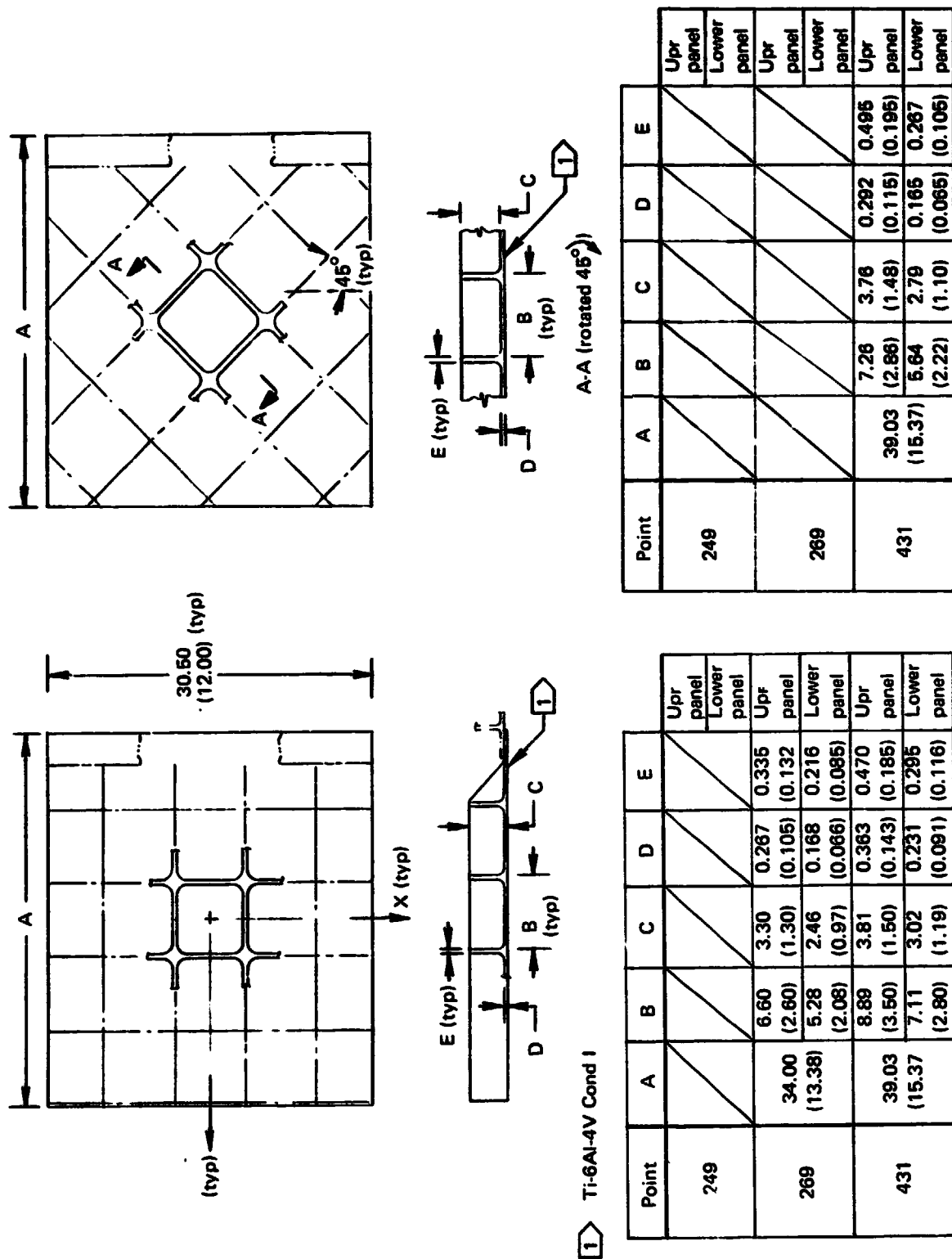
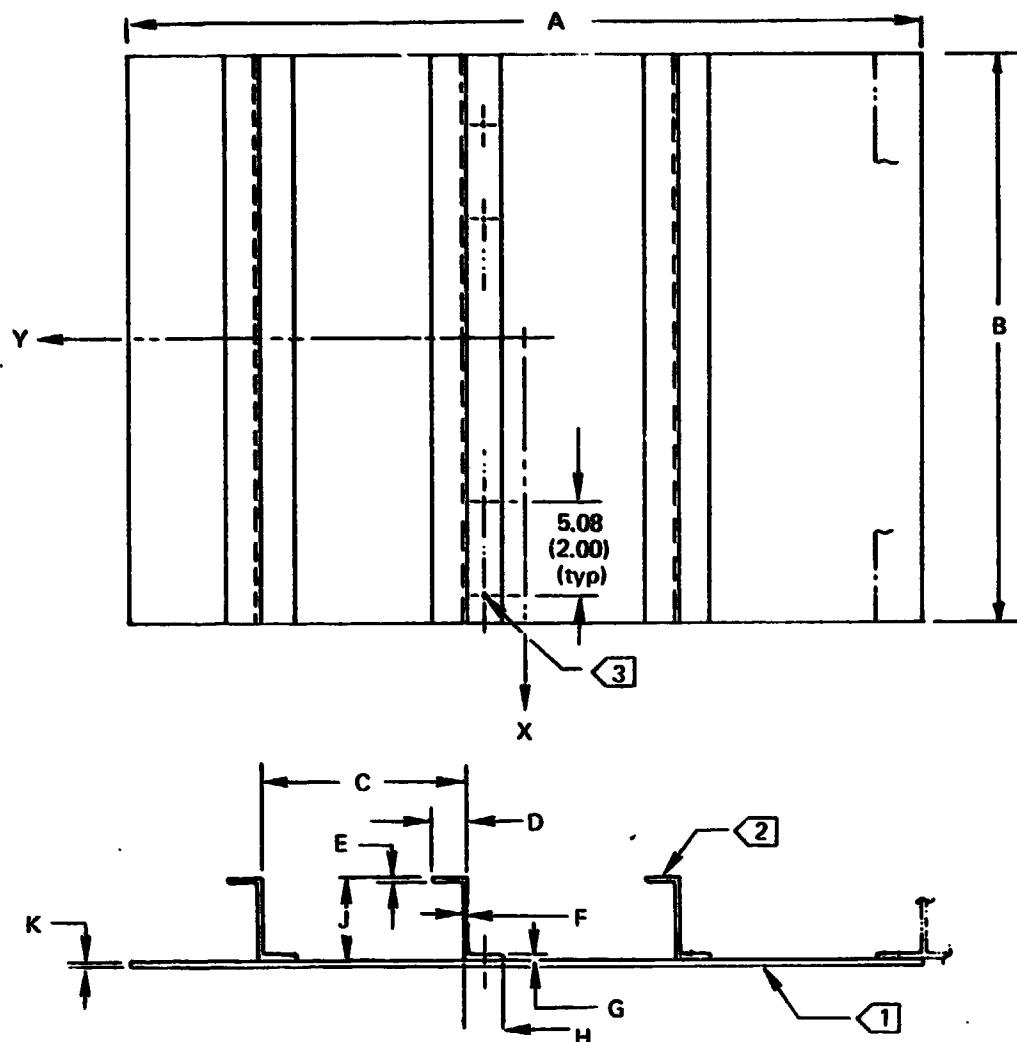


FIGURE 2-11.—WING SKIN PANEL—MACHINED WAFFLE



Point	A	B	C	D	E	F	G	H	J	K	
249											Upr Panel
											Lwr Panel
269	34.00 (13.38)	30.50 (12.00)	11.43 (4.50)	2.54 (1.00)	0.238 (0.094)	0.238 (0.094)	0.238 (0.094)	2.54 (1.00)	5.08 (2.00)	0.432 (0.170)	Upr Panel
			11.43 (4.50)	1.91 (0.75)	0.198 (0.078)	0.198 (0.078)	0.198 (0.078)	1.91 (0.75)	4.45 (1.75)	0.279 (0.110)	Lwr Panel
431	39.03 (15.37)	30.50 (12.00)	14.35 (5.65)	2.54 (1.00)	0.396 (0.156)	0.396 (0.156)	0.396 (0.156)	2.54 (1.00)	5.08 (2.00)	0.533 (0.210)	Upr Panel
			14.35 (5.65)	1.91 (0.75)	0.239 (0.094)	0.239 (0.094)	0.239 (0.094)	2.54 (1.00)	4.45 (1.75)	0.406 (0.160)	Lwr Panel

3 Ti slug rivets instl for high fatigue rate

2 Machined from 6-Al-4V extr

1 6Al-4V Cond I

See figure 2-1 for panel point locations
Basic dimensions—cm
() dimensions—in.

FIGURE 2-12.—WING SKIN PANEL—RIVETED SHEET STIFFENER

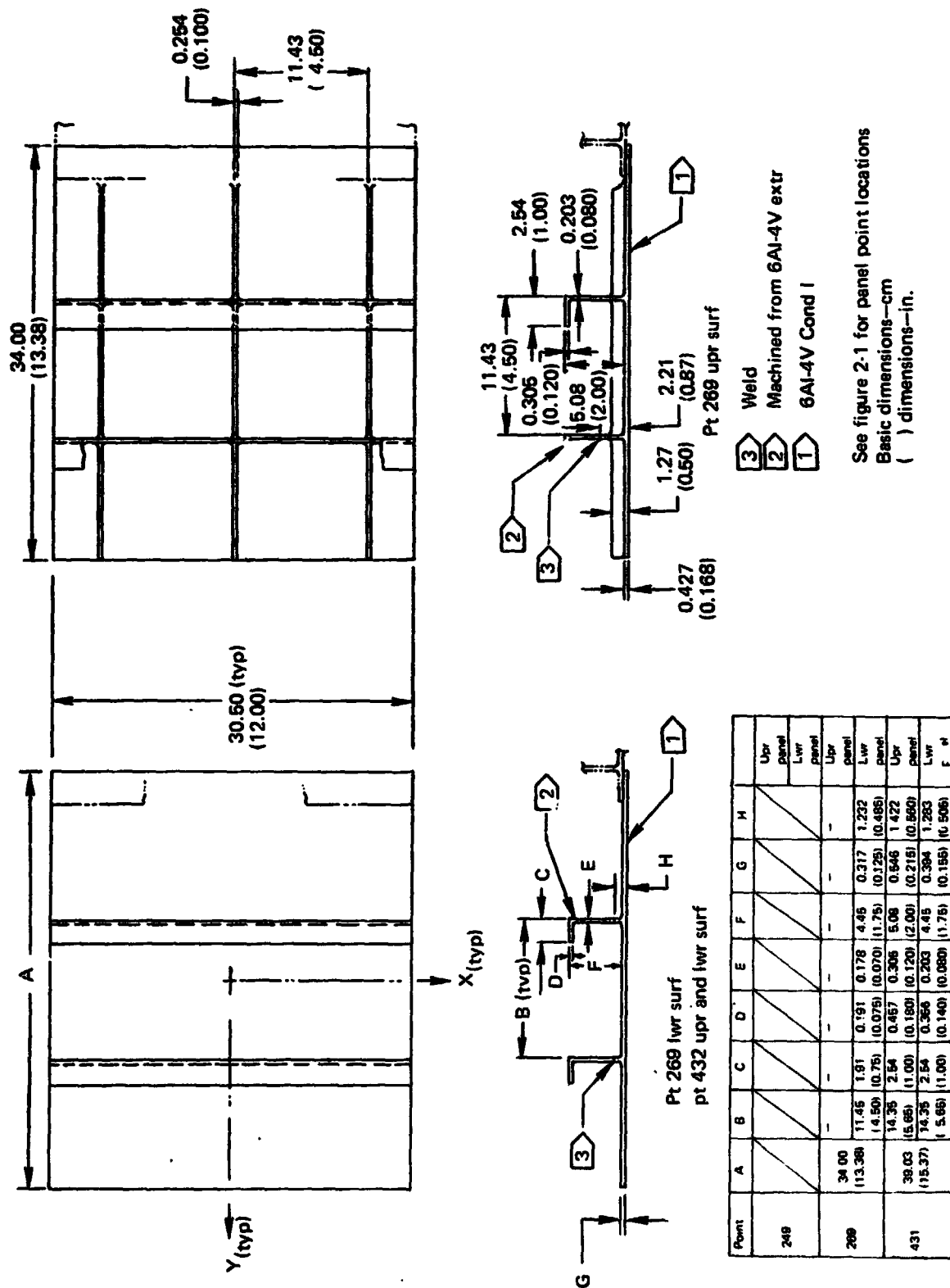
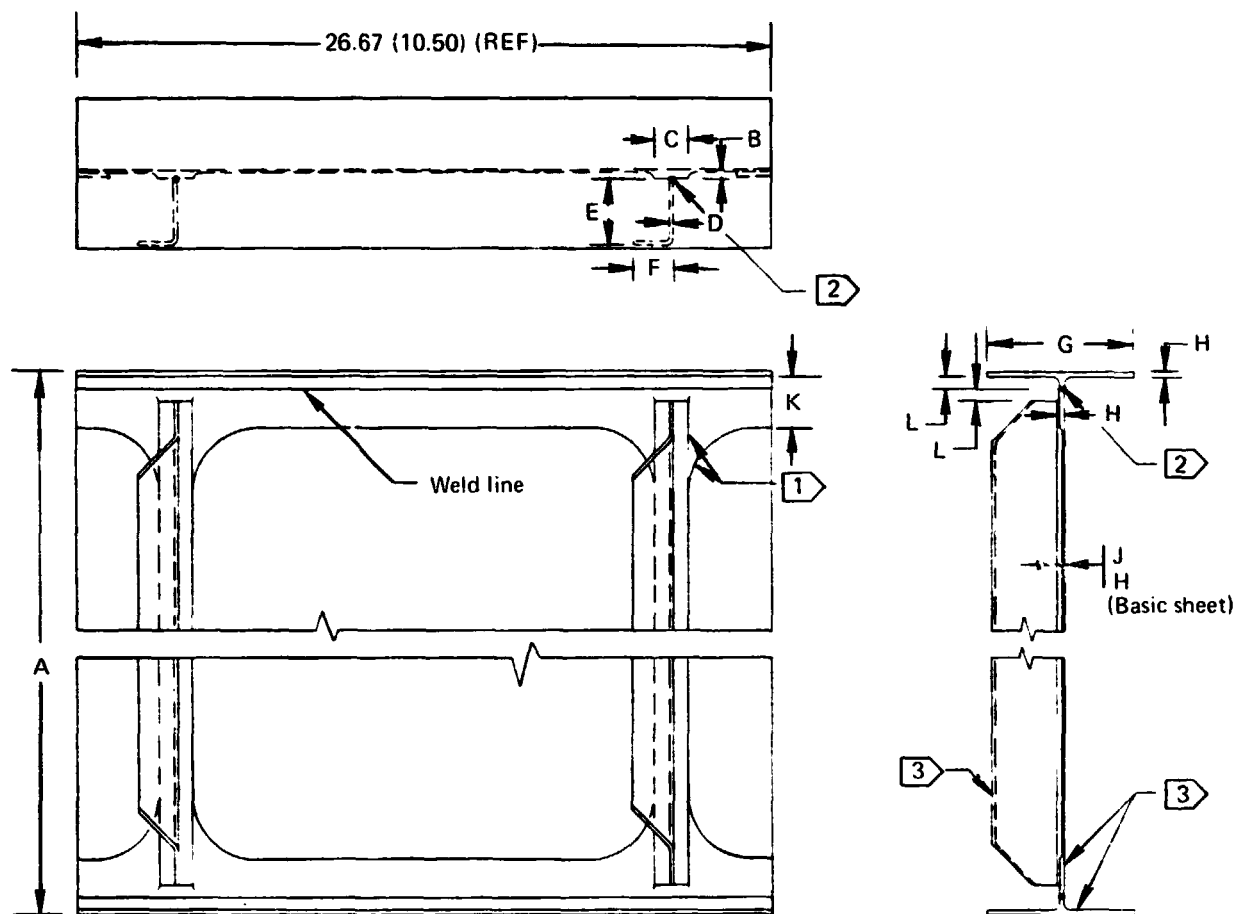


FIGURE 2-13.—WING SKIN PANEL—INTEGRALLY MACHINED AND WELDED—SHEET STIFFENER

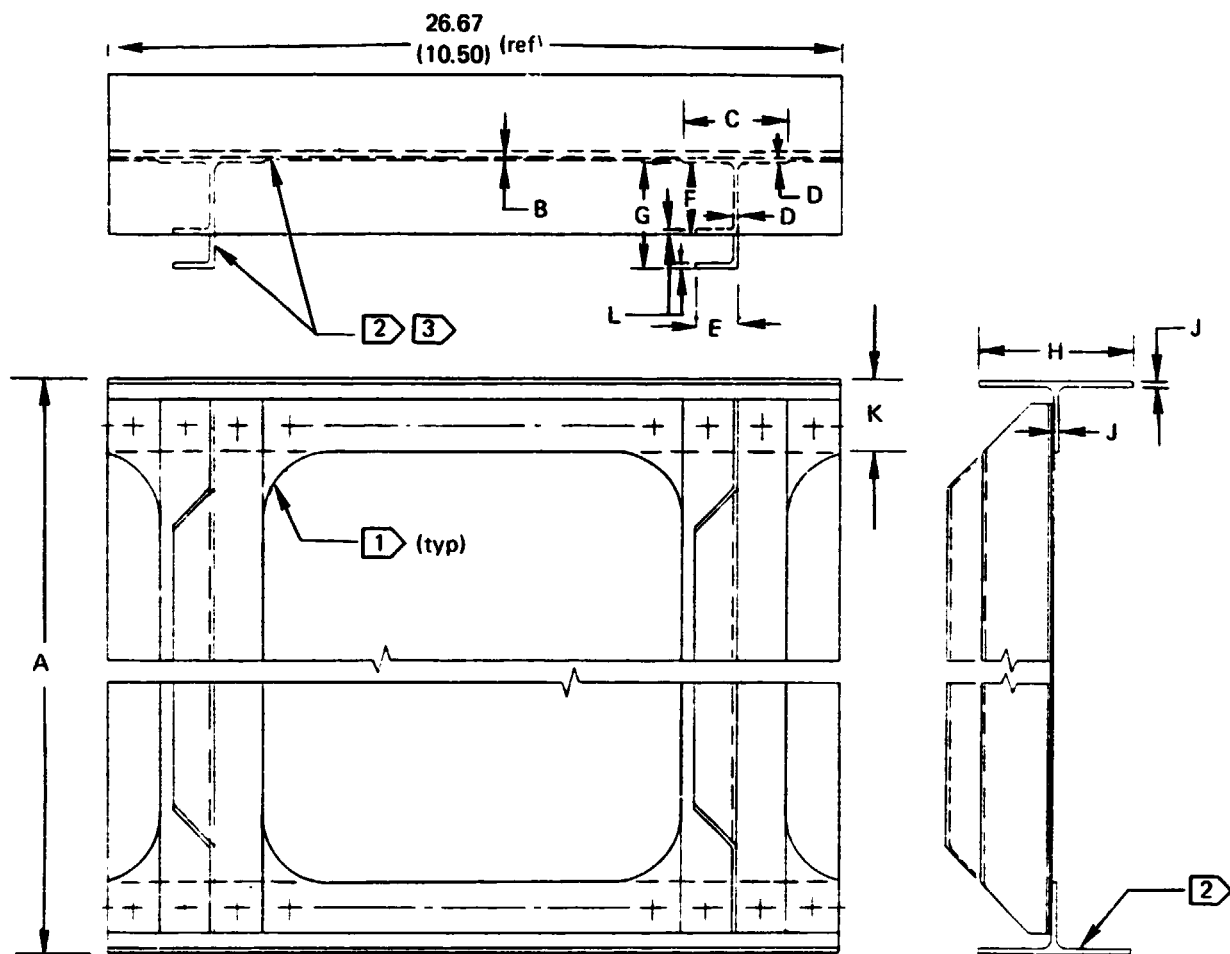


Basic dimension—cm
() dimensions—in.
See figure 2-1 for panel point locations

- 1 Step in chem-mill web
- 2 Fusion weld
- 3 Ti-6Al-4V Cond I

Point	A	B	C	D	E	F	G	H	J	K	L
249	83.82 (33.0)	0.279 (0.11)	1.27 (0.50)	0.178 (0.07)	3.81 (1.50)	1.52 (0.60)	5.59 (2.20)	0.203 (0.08)	0.038 (0.015)	1.91 (0.75)	0.457 (0.18)
269	58.42 (23.0)	0.279 (0.11)	1.27 (0.50)	0.140 (0.055)	3.81 (1.50)	1.52 (0.60)	5.59 (2.20)	0.203 (0.08)	0.038 (0.015)	1.91 (0.75)	0.457 (0.18)
431	33.02 (13.00)	0.279 (0.11)	1.27 (0.50)	0.102 (0.04)	2.54 (1.00)	1.52 (0.60)	5.59 (2.20)	0.203 (0.08)	0.038 (0.015)	1.91 (0.75)	0.457 (0.18)

FIGURE 2-14.—WING SPAR—WELDED SHEET STIFFENER



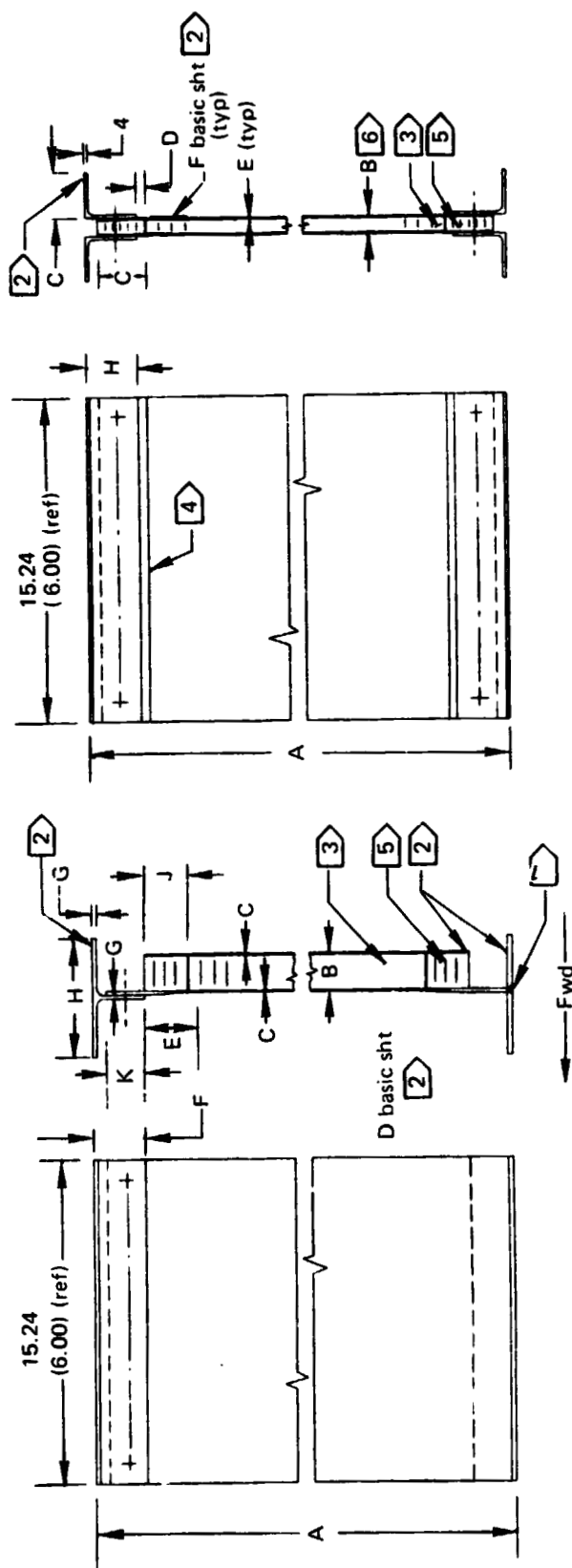
- 1 Step in chem-milled web
- 2 Ti-6Al-4V Cond I
- 3 Extruded web/stiffener panel
final shape as shown is obtained
through machining and subsequent
chem-milling

Basic dimensions—cm
() dimensions—in.

See figure 2-1 for panel point locations

Point	A	B	C	D	E	F	G	H	J	K	L
249	83.82 (33.0)	0.051 (0.020)	3.31 (1.50)	0.127 (0.050)	1.52 (0.60)		3.81 (1.50)	5.59 (2.20)	0.203 (0.080)	2.54 (1.00)	0.127 (0.050)
269	58.42 (23.0)	0.051 (0.020)	3.81 (1.50)	0.127 (0.050)	1.27 (0.50)	2.54 (1.00)		5.59 (2.20)	0.203 (0.080)	2.54 (1.00)	0.127 (0.050)
431	33.02 (13.0)	0.051 (0.020)	3.81 (1.50)	0.102 (0.040)	.127 (0.50)	2.54 (1.00)		5.59 (2.20)	0.203 (0.080)	2.54 (1.00)	0.102 (0.040)

FIGURE 2-15.—WING SPAR—MACHINED EXTRUSION—SHEET STIFFENER

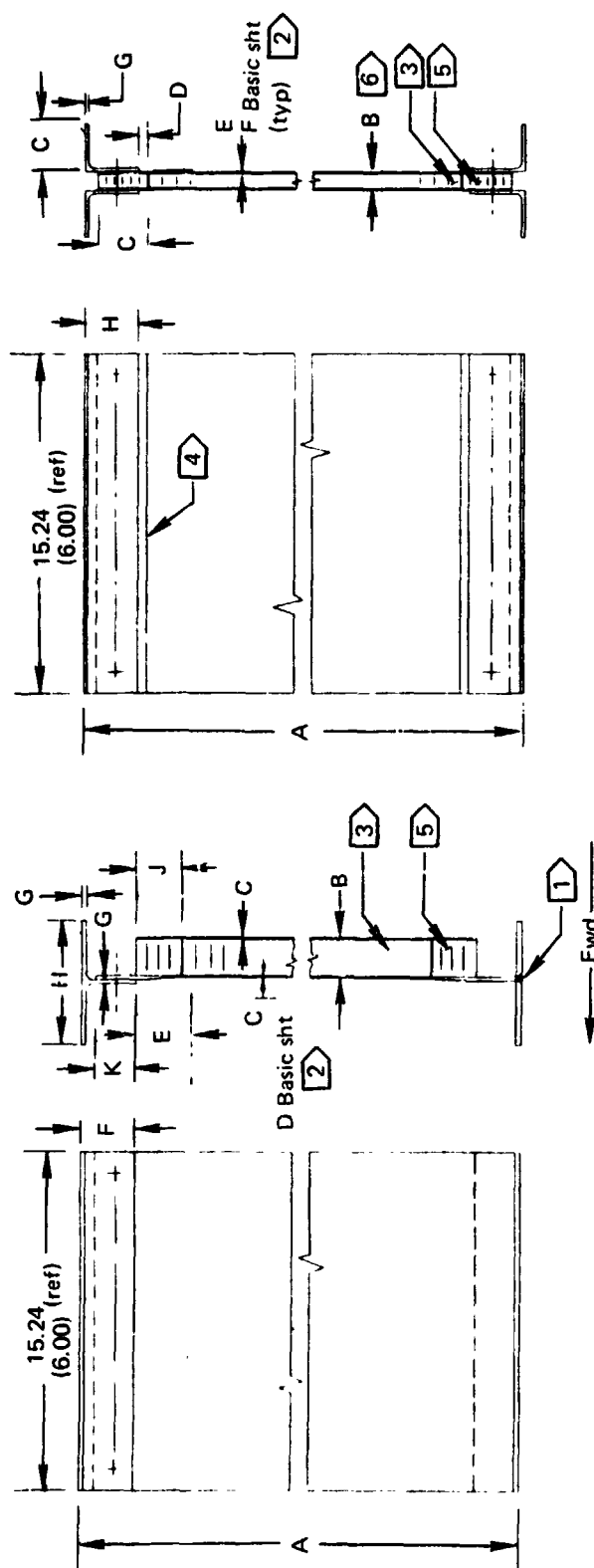


Point	A	B	C	D	E	F	G	H
249								
269	58.42 (23.00)	0.64 (0.25)	2.16 (0.85)	0.38 (0.15)	0.030 (0.012)	0.076 (0.030)	0.07 (0.050)	2.44 (0.96)
431	33.02 (13.00)	0.64 (0.25)	2.16 (0.85)	0.38 (0.15)	0.030 (0.012)	0.076 (0.030)	0.07 (0.050)	2.44 (0.96)

Point	A	B	C	D	E	F	G	H	J	K
249	83.82 (33.00)	1.65 (0.65)	0.030 (0.012)	0.360 (0.140)	3.30 (1.3)	2.64 (1.00)	0.203 (0.080)	5.59 (2.20)	2.54 (1.00)	1.78 (0.70)
269	58.42 (23.00)	0.64 (0.25)	0.030 (0.012)	0.254 (0.100)	2.29 (0.90)	2.64 (1.00)	0.203 (0.080)	5.59 (2.20)	1.78 (0.70)	1.78 (0.70)
431	33.02 (13.00)	0.64 (0.25)	0.030 (0.012)	0.178 (0.070)	1.52 (0.60)	2.64 (1.00)	0.203 (0.080)	5.59 (2.20)	1.27 (0.50)	1.78 (0.70)

- 1 Weld
- 2 Ti-6Al-4V Cond I
- 3 78.5 kg/m³ (4.9 pcf) SC4-20 core
Ti-3Al-2.5 V
- 4 Step in chem-milled web
- 5 224.3 kg/m³ (14.0 pcf) SS2-30 core
Ti-6Al-4V
- 6 This design used where core height
is 0.762 (0.30) or less

FIGURE 2-16.—WING SPAR—Al BRAZED TITANIUM HONEYCOMB



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Point	A	B	C	C	E	F	G	H
249								
269	58.42 (23.00)	0.76 (0.30)	2.16 (0.85)	0.38 (0.15)	0.020 (0.008)	0.076 (0.030)	0.102 (0.040)	2.44 (0.96)
431	33.02 (13.00)	0.76 (0.30)	2.16 (0.85)	0.38 (0.15)	0.020 (0.008)	0.076 (0.030)	0.102 (0.040)	2.44 (0.96)

Point	A	B	C	D	E	F	G	H	J	K
249	83.82 (33.00)	1.52 (0.60)	0.020 (0.008)	0.360 (0.140)	3.30 (1.30)	2.54 (1.00)	0.203 (0.080)	5.59 (2.20)	2.54 (1.00)	1.78 (0.70)
269	58.42 (23.00)	0.76 (0.30)	0.020 (0.008)	0.229 (0.090)	2.03 (0.80)	2.54 (1.00)	0.152 (0.060)	5.59 (2.20)	1.52 (0.60)	1.78 (0.70)
431	33.02 (13.00)	0.76 (0.30)	0.020 (0.008)	0.152 (0.060)	1.27 (0.50)	2.54 (1.00)	0.152 (0.060)	5.59 (2.20)	1.02 (0.40)	1.78 (0.70)

- 1 Weld
- 2 PH 15-7 mo
- 3 1/8 cell 68.9 kg/m³ (4.3 pcf) Ph 15-7 mo
- 4 Step in chem-mill weld
- 5 1/8 cell 365.9 kg/m³ (16.6 pcf) Ph 15-7 mo
- 6 This design used where core height is 0.762 (0.30) or less
- Basic dimensions—cm
() dimensions—in.
See figure 2-1 for panel
point locations

FIGURE 2-17.—WING SPAR—SILVER BRAZED STEEL HONEYCOMB

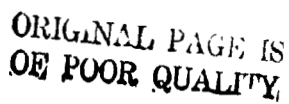


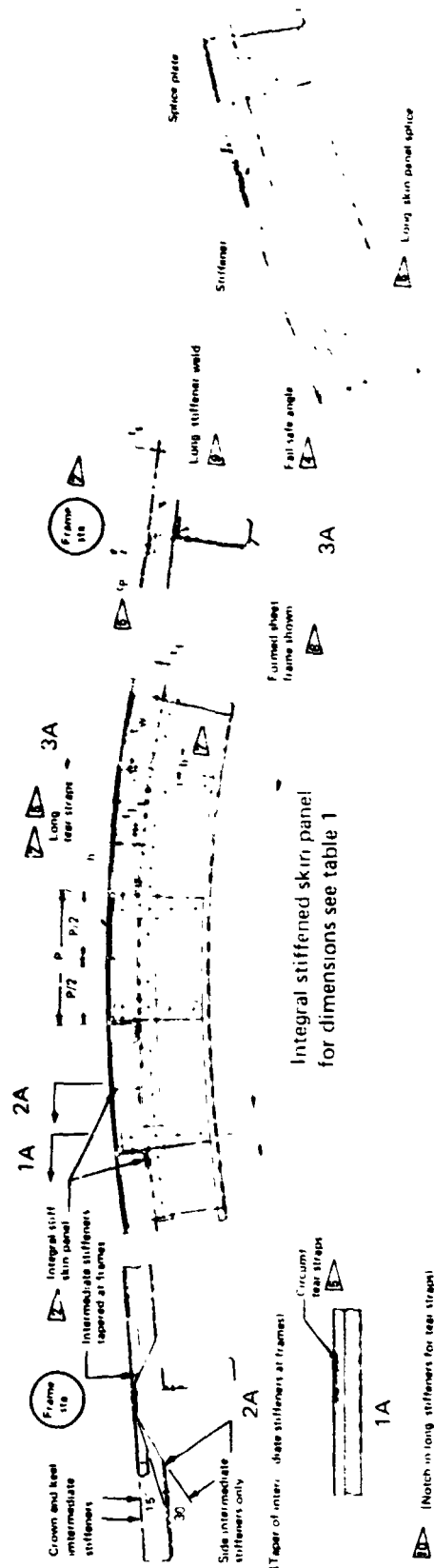
FIGURE 2-18.—BODY SKIN PANEL—Al BRAZED TITANIUM HONEYCOMB

Location pt no.	Body station cm (in.) Circumf location	H/C core depth cm (in.)	Outer face thickness cm (in.)	Inner face thickness cm (in.)	Effective pad up and splice matl  cm ² (in. ²)	Effective $\bar{t}$ cm (in.)	Panel load $\times 10^3$ N/m (lb/in.)	Panel weight
	Sta 2252.7 (Sta 1005.0)							
1	Upr crown 	0.95 (0.375)	0.041 (0.016)	0.025 (0.010)	3 splices 19.35 (3.00)	0.163 (0.064)	11,064 (2,485) ten 8,845 (1,988) comp	
2	Side 	0.95 (0.375)	0.041 (0.016)	0.025 (0.010)	—	0.066 (0.026)	1,060 (605) shear	
	Sta 4368.8 (Sta 1720.0)							
3	Upr crown 	1.27 (0.50)	0.074 (0.029)	0.074 (0.029)	3 splices 19.35 (3.00)	0.223 (0.088)	29,580 (6,646) ten 24,791 (5,570) comp	
4	Side 	1.91 (0.75)	0.043 (0.017)	0.043 (0.017)	—	0.086 (0.034)	1,560 (890) shear	
5	Lwr crown 	2.54 (1.00)	0.108 (0.042)	0.108 (0.042)	3 splices 19.35 (3.00)	0.302 (0.119)	52,985 (11,921) comp 44,407 (9,991) ten	
	Sta 6283.9 (Sta 2477.0)							
6	Upr crown 	1.91 (0.75)	0.182 (0.066)	0.182 (0.066)	5 splices 51.60 (8.00)	0.554 (0.218)	122,622 (27,564) ten 38,911 (8,750) comp	
7	Side 3 Sta 7874.0 (Sta 3100)	3.17 (1.25)	0.059 (0.023)	0.059 (0.023)	—	0.118 (0.046)	6,966 (3,980) shear	
8	Upr crown 	1.91 (0.75)	0.041 (0.016)	0.025 (0.010)	5 splices 40.30 (6.25)	24.15 (9.50)	60,317 (13,570) ten 49,615 (11,160) comp	
9	Side 	2.54 (1.00)	0.041 (0.016)	0.025 (0.010)	—	0.066 (0.026)	1,219 (1,210) shear	
10	Lwr crown 	3.17 (1.25)	0.132 (0.052)	0.132 (0.052)	3 splices 24.18 (3.75)	22.55 (8.88)	69,501 (15,627) comp 57,169 (12,852) ten	

FIGURE 2-18.—BODY SKIN PANEL—AI BRAZED TITANIUM HONEYCOMB (Continued)

- 15 Frame and floor beam spacing is double that of the baseline concept figure 2-7 because of the improved stability of the sandwich shell.
- 14 Potential 1980 design alternative for minimum weight structure, providing 10 11 12 have been successfully developed.
- 13 This criteria may be waived by 1975 or 1980 because of technological advancement.
- 12 Faying surface brazing is beyond current technology but is under development and may be available for 1975.
- 11 The location of the brazed frame chord on the panel is critical for splice and load continuity. Current technology cannot maintain acceptable location tolerances.
- 10 Current mfg criteria allow for only a single step in core machining in a given panel because of the extreme sensitivity of the braze process to fit up tolerances. 13
- 9 Circumferential tear strap matl is not included in effective $\bar{t}$.
- 8 Longitudinal tear strap matl is included in effective $\bar{t}$.
- 7 Frame, chord and stiffener matl is Ti-6Al-4V Cond I.
- 6 448 kg/m³ (28 pcf) h/c core SS2-60 commercially pure Ti. to be used with 0.635 cm (1/4 in.) dia fasteners.
- 5 224 kg/m³ (14 pcf) h/c core SS2-30 commercially pure Ti. to be used with 0.467 cm (3/16 in.) dia fasteners.
- 4 57.7 kg/m³ (3.6 pcf) h/c core SC4-20 commercially pure Ti.
- 3 Skin matl is Ti-6Al-4V Cond I.
- 2 Skin matl is Ti-6Al-4V Cond I.
- 1 Panel $\bar{t}$ shown is in T. 8

FIGURE 2-18.—BODY SKIN PANEL—AI BRAZED TITANIUM HONEYCOMB (Concluded)



Note:

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- 3
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For design conditions, loads, body characteristics, and load point locations, figure 2-7

Integral stiffened skin panel Ti-6Al-4V Cond. I

Chem-milled Titanium formed sheet frame depicted as typical only.

Frame fail safe chord

Skin pad up for additional frame chord material, minimum fastener thickness requirements, tear straps, or splice plates not included in $\bar{t}$.

The basic international unit system of measurement (SI) is used in the table; U.S. customary decimal unit values are in parentheses

Frame spacing 43.82 cm (17.25 in.)

Panel " $\bar{t}$ " quoted, is midway between frames

Weld

Circumferential tear straps located midway between frames

FIGURE 2-19—BODY SKIN PANEL—INTEGRALLY MACHINED SHEET STIFFENER

	Location point number	Circumf load point location	Panel t cm (in.)	Skin t _s cm (in.)	Web t _w cm (in.)	Flange t _f cm (in.)	Flange width b cm (in.)	Stiff height h cm (in.)	Stiff height h cm (in.)	Stiff t cm (in.)	Stiff pitch p cm (in.)	Design cond.	Panel weight kg/cm ² (lb/in. ²)
Sta 2552.70 (Sta 1005.00)	1	Upr crown	0.130 (0.051)	0.091 (0.036)	0.094 (0.037)	0.112 (0.044)	1.524 (0.600)	2.845 (1.120)	1.143 (0.450)	0.094 (0.037)	13.564 (5.340)	B	
	2	Side 9	0.168 (0.066)	0.097 (0.038)	0.127 (0.050)	0.152 (0.060)	1.270 (0.500)	2.540 (1.000)	-	-	13.564 (5.340)	A	
Sta 4367.80 (Sta 1720.00)	3	Upr crown	0.234 (0.092)	0.137 (0.054)	0.152 (0.060)	0.229 (0.090)	2.540 (1.000)	3.327 (1.310)	1.194 (0.470)	0.152 (0.060)	12.446 (4.900)	B	
	4	Side 9	0.262 (0.103)	0.107 (0.042)	0.132 (0.052)	0.229 (0.090)	2.540 (1.000)	3.327 (1.310)	-	-	12.446 (4.900)	A	
	5	Lwr keel	0.377 (0.147)	0.229 (0.090)	0.178 (0.070)	0.381 (0.150)	2.540 (1.000)	3.810 (1.500)	1.397 (0.550)	0.330 (0.130)	12.446 (4.900)	A	
Sta 6283.90 (Sta 2474.00)	6	Upr crown	0.516 (0.203)	0.277 (0.109)	0.356 (0.140)	0.483 (0.190)	3.175 (1.250)	3.810 (1.500)	1.143 (0.450)	0.356 (0.140)	11.862 (4.670)	A	
	7	Side 9	0.429 (0.169)	0.170 (0.067)	0.211 (0.083)	0.330 (0.130)	2.540 (1.000)	3.810 (1.500)	-	-	11.862 (4.670)	A	
Sta 7874.00 (Sta 3100.00)	8	Upr crown	0.338 (0.133)	0.201 (0.079)	0.178 (0.070)	0.279 (0.110)	2.540 (1.000)	3.327 (1.310)	1.270 (0.500)	0.279 (0.110)	10.973 (4.320)	B	
	9	Side 9	0.314 (0.124)	0.124 (0.049)	0.178 (0.070)	0.203 (0.080)	2.540 (1.000)	3.327 (1.310)	-	-	10.973 (4.320)	A	
	10	Lwr keel	0.424 (0.167)	0.211 (0.083)	0.229 (0.090)	0.406 (0.160)	3.175 (1.250)	3.810 (1.500)	1.397 (0.550)	0.284 (0.112)	10.973 (4.320)	A	

Figure 2-19. - BODY SKIN PANEL - INTEGRALLY MACHINED SHEET STIFFENER (CONCLUDED)

**Corrugated Core Skin Panel
For Dimensions See Table**

For design conditions, loads, body characteristics, and node load point locations, see Figure 2-7
Construction materials Ti-6Al-4V, Cond 1
Chem-milled, formed sheet frame depicted as typical only
The basic international unit system of measurement (SI) is used in table, english equivalent unit values are in parenthesis

1  Frame spacing 87.6 cm (34.50 in.) approx

FIGURE 2-20.—BODY SKIN PANEL—CORRUGATED CORE SANDWICH

Location point number	Circumf node pt location	Panel $\bar{r}$ cm (in.)	Skin t_s cm (in.)	Core t_w cm (in.)	Panel weight n/cm (lb/in.)
1	Upr crown				
		0.193 (0.076)	0.064 (0.025)	0.038 (0.015)	
2	Side	0.193 (0.076)	0.064 (0.025)	0.038 (0.015)	
3	Upr crown				
		0.267 (0.105)	0.089 (0.035)	0.053 (0.021)	
4	Side	0.193 (0.076)	0.064 (0.025)	0.053 (0.015)	
5	Lowr keel	0.432 (0.170)	0.142 (0.056)	0.086 (0.034)	
6	Upr crown	0.521 (0.205)	0.173 (0.068)	0.104 (0.041)	
7	Side	0.254 (0.100)	0.084 (0.033)	0.051 (0.020)	
8	Upr crown				
		0.136 (0.053)	0.045 (0.018)	0.027 (0.011)	
9	Side	0.193 (0.076)	0.064 (0.025)	0.038 (0.015)	
10	Lowr keel	0.381 (0.150)	0.127 (0.050)	0.076 (0.030)	

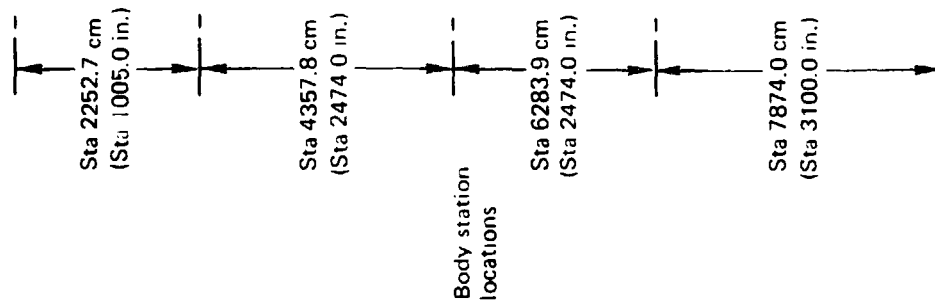
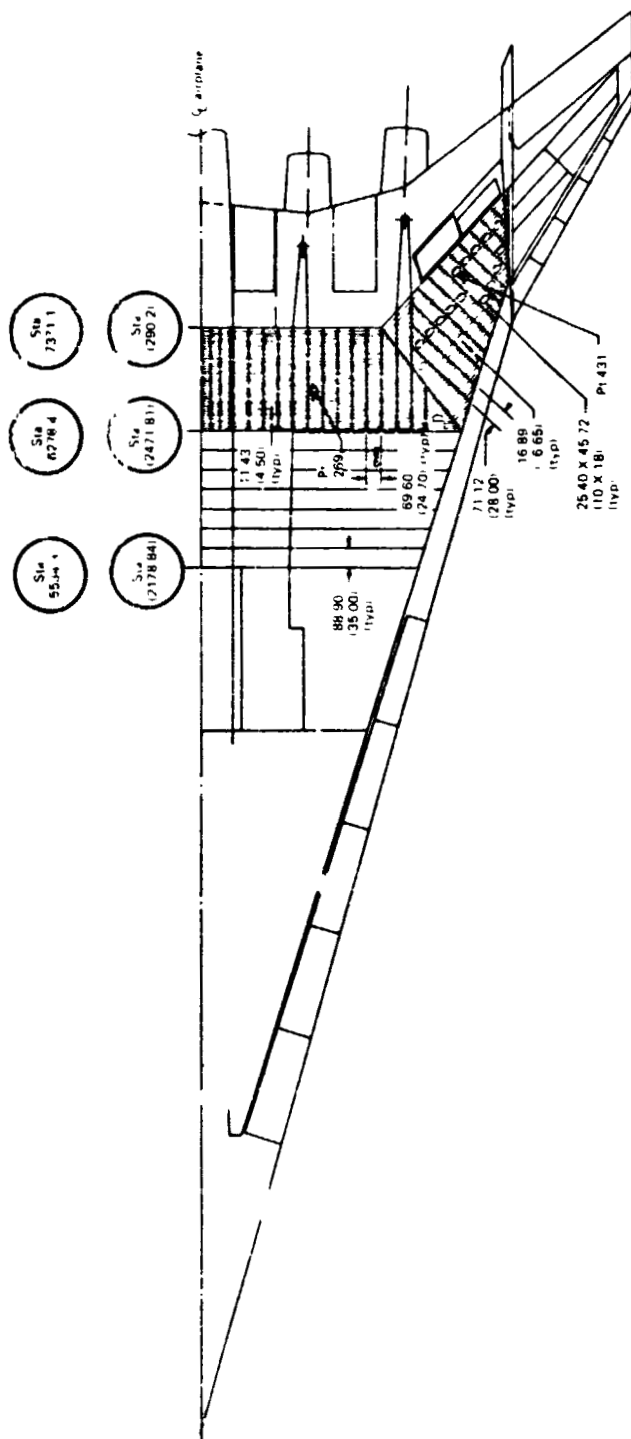
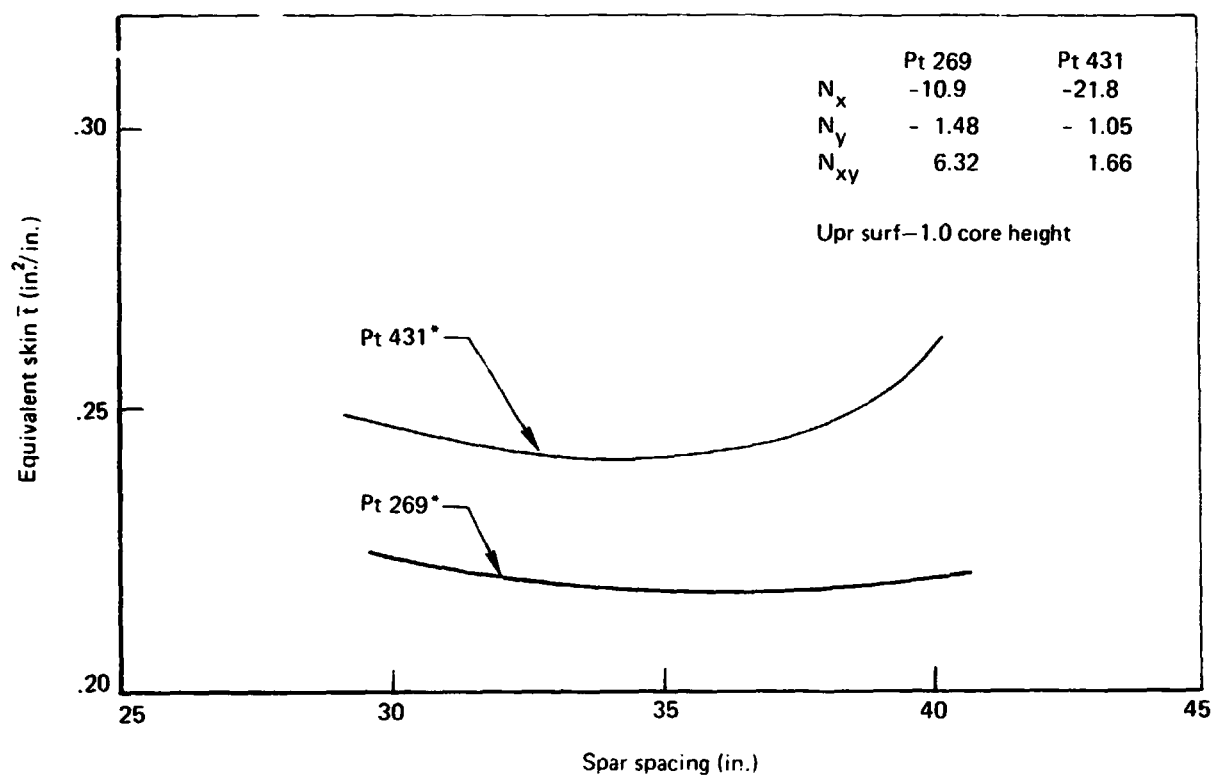


FIGURE 2-20.—BODY SKIN PANEL—CORRUGATED CORE SANDWICH (Concluded)



Basic dimensions—cm
() dimensions—in.

FIGURE 2-21.—WING STRUCTURAL ARRANGEMENT—SHEET STIFFENER



*Equiv $\bar{t}$ includes—skins, edge member and spar chord

FIGURE 2-22.—WING SPAR SPACING

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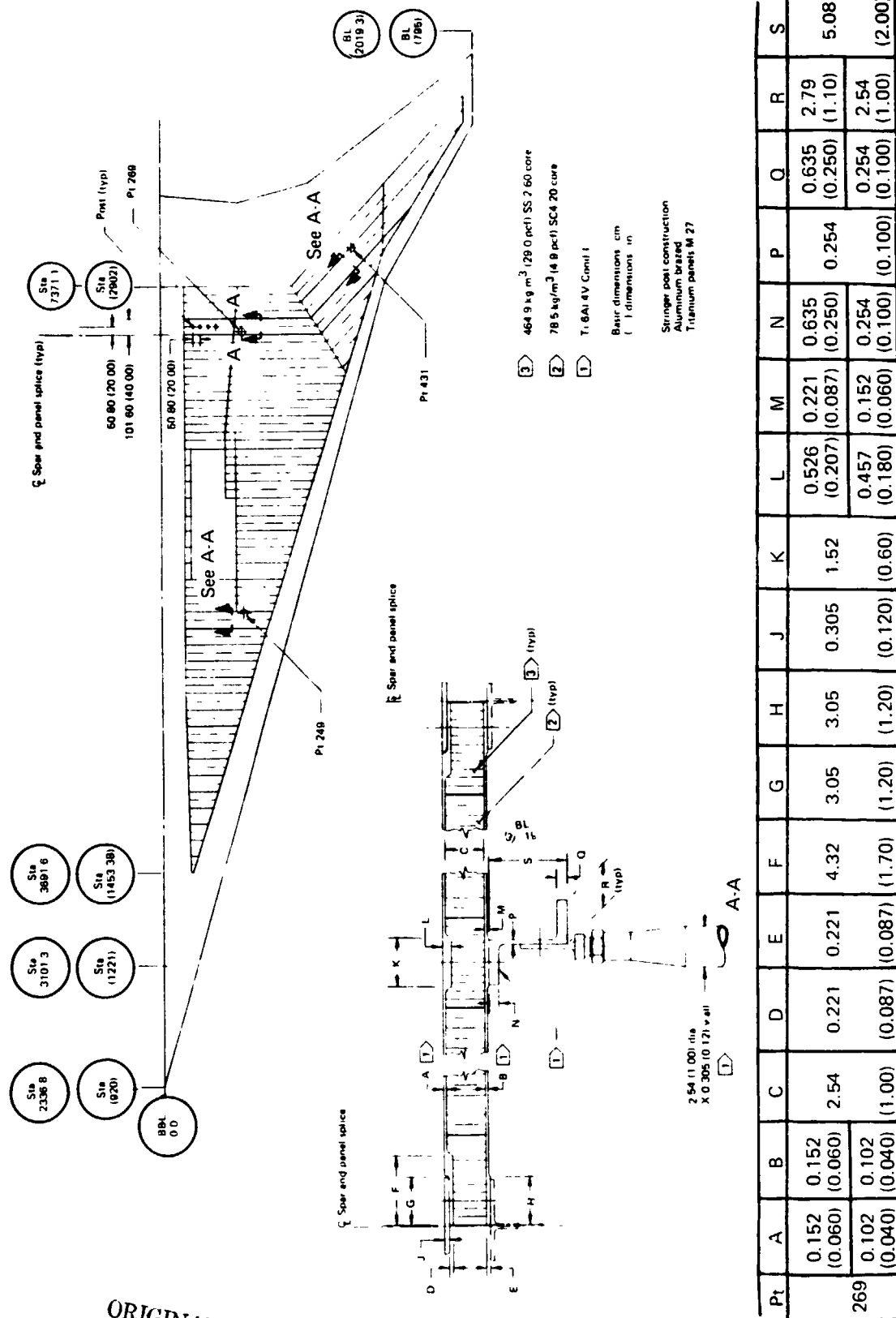


FIGURE 2-23.—POST AND SPAR STRUCTURAL ARRANGEMENT

SECTION 3

MATERIALS EVALUATION AND SELECTION

by

C. L. Hendricks

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SECTION 3

MATERIALS EVALUATION AND SELECTION

MATERIALS FOR 1975 AND MACH 2.7

Recommendations for Mach 2.7, based upon 1975 technology, are restricted to materials and processes that are virtually proven for primary airframe applications because of the short time remaining for development. The materials are divided into three main categories--Metals, Advanced Composites Adhesive Bonding, and Fuel Tank Sealant Insulation. Materials for various applications were finally selected by screening of potential candidates, considering a combination of availability, environmental effects, production capability, cost, and projected status of specifications and allowables. The chart in figure 3-1 illustrates this selection process.

The following paragraphs describe the selected materials.

METALS

Titanium

Ti-6Al-4V has been modified and improved during the National SST program and the subsequent Department of Transportation (DOT) funded technology follow-on program. The improved Ti-6Al-4V beta processed material possesses superior fracture and fatigue properties. These improved properties, however, are achieved at the expense of a 3 to 5% strength loss and 7 to 12% higher cost. Specifications for this improved Ti-6Al-4V have been written and are designated Advanced Supersonic Technology (AST) specifications. The AST Ti-6Al-4V material is recommended for high toughness and fatigue design applications, whereas conventional MIL specification materials are recommended for airframe components designed primarily by strength or stiffness.

Specific recommendations for strength and stiffness designed components are as follows:

- Sheet (thicknesses .016 to .187 inches) Ti 6Al-4V annealed per MIL-T-9046;
- Plate (thicknesses .188 to 4.0 inches) Ti 6Al-4V annealed per MIL-T-9046; where component geometry minimizes quench distortion, higher strength can be gained using Ti 6Al-4V STA (Solution Treated and Aged) per MIL-T-9046;
- Bar and Forgings (thicknesses less than 1.25 inches) Ti 6Al-4V STA per MIL-T-9047;
- Bar and Forgings (thicknesses 1.25 through 2.00 inches) Ti 6Al-6V-2Sn STA per MIL-T-9047;
- Extrusions - Ti 6Al-4V Annealed or STA per BMS 7-44.

Design allowables for the above materials are available from sources such as the Boeing Design Manual.

The following materials are recommended for fracture and fatigue design applications:

- Sheet (Thicknesses .016 to .187 inches) Ti 6Al-4V Annealed per AST-1.
- Plate (thicknesses .187 to 4.00 inches) Ti 6Al-4V Annealed per AST-2.
- Bar and Forging - Ti 6Al-4V Annealed per AST-3
- Extrusions - Ti 6Al-4V Annealed per AST-4.

Design allowables need to be developed for the AST specification materials. In the interim, until such allowables are available, SST allowables for material purchased to XBMS 7- 174B should be used.

In addition to design allowables, research is required in heat treated titanium alloys to correct dimensional tolerance problems caused by water quenching. Depth hardening, another problem, is presently limited to about 1.5 inch thicknesses.

Steel

The following steels are recommended for design:

- Sheet and Plate PH 15-7 Mo (Th 1050 heat treat)
- Forgings 15-5 PH corrosion resistant steel and Custom 455 corrosion resistant steel for hot structure. 300M low alloy steel for cooled structure such as landing gear.

ADVANCED COMPOSITES/ADHESIVES

Advanced composites research and development has been actively supported by government and industry funding for the past several years. However, only a very limited amount of the available data is considered applicable to Mach 2.7 commercial SST technology. None of the high modulus, high strength fibrous composites are developed sufficiently to be considered ready for primary structural applications which are subjected to 50,000 hour life and -65° to 450° F thermal cycling. Much of the work to apply advanced composites to airframe structures has been restricted to short life military applications of 100 to 1,000 hours and higher temperatures, ranging from 450° to 600° F. Figures 3-2 through 3-4 illustrate the effect of time-temperature exposure on epoxy and polyimide composites based on very limited test data.

The major technical problems remaining to be resolved for composites are:

- Material and process reliability sufficient to produce quality hardware for airframe structure.

- Design confidence from allowables and simulated and actual flight cycle testing.
- Methods for efficient composite/metal joint attachment to fully utilize composite properties.
- In service reliability, maintenance, and repair experience to assure fabrication of relatively trouble free structure.

The composite combinations described below, which are considered to be candidate SST materials, are subject to specific technical limitations which prevent their utilization in airframe structure by 1975.

Borsic/Aluminum (Bsc/Al)-Honeycomb sandwich face skins with titanium honeycomb core still require a less degrading braze process. The major portion of work completed to date relies on a 1-2 minute dip-braze process at 1100° F on relatively small parts (ref. 3-1). This process lacks necessary time/temperature control to prevent Bsc filament damage in fabrication of production size panels. The development of a 900° braze alloy process to eliminate filament damage plus development of allowables test data does not appear likely for 1975 production.

Sheet-stiffener construction utilizing titanium skins and Bsc/Al reinforcement fabricated by compacting the composite against the titanium may be a feasible process by 1975 depending upon the availability of development funds.

Boron/Polyimide (B/PI) and Graphite/Polyimide (Gr/PI) are the high temperature organic matrix composites on which most development and evaluation has been conducted. These materials lack the basic technical information needed to be considered for sandwich face skins for Mach 2.7, 450° F by 1975.

Additional development required to assure confidence with polyimide or other high temperature composite designs is listed below:

- Generation of high temperature stability and thermal cycling data combined with simulated stress and fatigue properties to demonstrate long time 450° F service capability.
- Development of adhesive and related surface preparation techniques and supporting allowables for composite bonded assemblies.
- Development of non-porous thermally stable polymers which can be processed into production size parts or development of a seal coating system for existing condensation reaction heat stable polymers.
- Development of techniques for reduction of residual stress caused by different coefficients of thermal expansion in metal/composite bonded assemblies under production operations (ref. 3-2).

Newer polymeric systems, such as polyphenylquinoxaline (PPQ) and polyimidazoquinazoline (PIQ) which exhibit improved thermal stability will not be ready for 1975 production requirements.

Adhesive bonded metal sandwich assemblies will not be ready for commercial SST use at Mach 2.7, 450° F, by 1975. The most thermally stable polyimide adhesive systems generate condensation-reaction volatiles resulting in a porous bond line. Metal surface preparation techniques which produce adequate bond strengths with high reliability remain to be developed. The composite systems, although not presently proven for Mach 2.7 primary structural application, offer significant potential improvement in airframe efficiency in the 1980+ time period. Continued development, evaluation, and generation of data in the problem areas previously discussed are necessary to assure timely utilization of these materials in future airplane design.

FUEL TANK SEALANT

The only fuel tank sealant in a stage of development and testing sufficient to provide confidence in its reliability is a fluorosilicone, DC-94-529, marketed by Dow Corning. It had been selected for use in the SST prototype, based on screening tests of fourteen candidate materials. It has been tested extensively under a contract with the Department of Transportation. Test specimens are being exposed to two cyclic environments similar to the conditions experienced in flight. One test sequence duplicates a typical flight and the other imposes the most severe high temperature, i.e., 440° F, and fuel vapor environment. A third test utilizes a small sealed tank exposed to alternating cycles of environmental exposure and loading. The tank sealant has functioned satisfactorily after 10,000 hours of the high temperature exposure, although numerous small splits are evident. Figure 3-5 shows the test installation used for the tank test mentioned above.

Studies have investigated DC-94-529 adhesion to titanium, compatibility to other materials, reversion resistance and repairability. Unreliable adhesion to titanium and low strength and elongation after elevated temperature exposure are recognized deficiencies. There is no suitable injection or faying surface sealant for a Mach 2.7 SST.

FUEL TANK INSULATION

Insulation is needed for two purposes: the systems lines routed through the fuel tanks must be insulated to prevent heat transfer to the fuel, and the tank walls and structure must be insulated to prevent premature fuel vaporization. The amount of insulation needed depends on the net conductance of the fuel tank structure. Figure 3-6 shows the conductance of aluminum brazed honeycomb sandwich panels, and figure 3-7 shows the net tank conductance as a function of panel conductance. Figure 3-8 presents the materials considered for fuel tank insulation.

Conduit insulation has been developed and tested, but not optimized. The most practicable type is a foamed elastomeric fluorosilicone which may be slipped or slit and wrapped over the tubing, and joints sealed with the fluorosilicone fuel tank sealant. The material has been made in a density of 20 and 40 pounds per cubic foot. Close density

control has therefore been impossible. Both densities functioned favorably in simulated flight testing. Thermal conductivity of the 40 lb/cubic ft material was measured at 0.4 Btu-in ft²hr⁻¹F. Thermal conductivity of the 20 lb/cubic ft material was estimated to be 0.3 Btu-in ft²hr⁻¹F. Further development of materials and methods can be expected to result in a reduction of both density and thermal conductivity by one-half.

The type and extent of insulation for tank structure will depend upon structural design. Where possible, sealed evacuated batts may be used consisting of titanium envelopes filled with dextraglass having a conductivity of 0.04 Btu-in ft²hr⁻¹F and a density of 24 to 48 lbs cubic ft. If vacuum is lost, the K value increases to .4 Btu-in ft²hr⁻¹F. The batts may be tack welded in place or bonded with fuel tank sealant. This concept is in an advanced stage of development.

A flexible foam insulation may be used for areas that are irregular in contour or too small for batts to be advantageous. It could also be used to cover stiffeners and other extensions into the fuel. A density as low as 17 lbs cubic ft is attainable with a K of .3 Btu-in ft²hr⁻¹F. It can be attached to surfaces with fuel tank sealant. Further development and testing are necessary to define controls needed to make it reproducible and eliminate the tendency to cause stress corrosion in titanium.

MATERIALS FOR 1980+ AND MACH 2.7

Material recommendations for the Mach 2.7 Arrow Wing SST based upon 1980+ technology assume completion of required developmental work and generation of structural design allowables. The three main areas of metals, composites adhesives, and fuel tank sealant/insulations are discussed individually. Figure 3-9 presents the 1980+ material selection chart.

METALS

Evaluation of candidate materials indicates that there are no unreinforced metallic materials likely to compete with titanium alloys. In order to be competitive, ferrous alloys would need a large increase in strength with an associated improvement in toughness. These two properties in the past have proven to be mutually exclusive. Similar comments are valid for the super alloys. The lower density materials, such as Beryllium, are neither strong nor tough enough. There appear to be no major improvements available in high temperature performance of the aluminum alloys. Thus the main requirement is to develop improved titanium alloys. For damage tolerant structural applications the beta processed ASTM specifications for 6Al-4V still appear to be the best selection.

New titanium alloys with higher strength properties are being developed which air harden in thicknesses up to 6 inches. These are designated Ti 17, Ti 6-2-2-2-2, Beta C and Ti 6-2-46.

Ultra high strength steel, 300M still appears to be the best alloy selection in the 275-300 ksi range. A corrosion resistant alloy, AFC 77B, which was being developed for use in this strength range does not have acceptable toughness and stress corrosion resistance.

For high strength forgings, the corrosion resistant alloys 15-5PH and Custom 455 continue to look attractive. MP 35N is a fairly recent super alloy development which may be cold worked to develop high strengths. It has excellent stress corrosion resistance and could develop into an excellent structural material. It is currently finding major use as a fastener material.

ADVANCED COMPOSITE/ADHESIVES

Advanced composites and adhesive bonding are the base from which future airplanes can derive increased structural efficiency and performance. The application of these materials to primary airframe structures designed in 1980 will depend directly upon the extent of development, and related allowables and test data that are available at that time. Such information must include data specific to a commercial supersonic airplane to be of benefit to the aircraft industry.

The following technical problems in composite materials require solution prior to use in primary airframe structures:

1. Development of sufficient simulated and actual flight service environmental testing specific to commercial SST requirements and design allowables necessary to support design and production.
2. Determination of optimum design for joining composites and metals, mechanical joints, and new concepts in joint configuration.
3. Development of titanium and composite surface preparation for adhesive bonding in reinforced metal and bonded joint structures.
4. Determination of basic mechanisms of failure induced by thermal cycling and fatigue in all-composite structure and in the bond line of composite reinforced metal structure.
5. Determination of fracture toughness properties of all-composite structure.
6. Preparation of material and process specifications to present standards to support fabrication of commercial airplane structural components.
7. Development of in-service reliability, maintenance, and repair procedures.

Additional comments specific to each candidate composite system related to required developmental effort and potential and existing problems are discussed below.

The composite material systems most likely to be utilized in primary structure in the 1980+ time span for a Mach 2.7 SST are:

- Borsic/Aluminum (Bsc/Al)
- Borsic/Titanium (Bsc/Ti)
- Boron/Polyimide (B/PI)
- Boron/Polyphenylquinoxaline (B/PPQ)
- Graphite/Polyimide (G/PI)
- Graphite/Polyphenylquinoxaline (G/PPQ)

Bsc/Al will be available as single layer unidirectional tape for compaction to laminate in tailored orientation or as laminated sheet sections. The sheet sections can be hot formed parallel to the fiber axis in unidirectional laminates for hat or zee configurations and brazed to titanium and/or Bsc/Al sheets for stiffened wing and body panels. Reference 3-3 and figure 3-10 provide allowables for Bsc/Al laminate. Other methods of joining which may be used are resistance spot and seam welding, autoclave diffusion bonding, and step-press braze/bonding.

The areas of work which remain, in addition to those listed above, are development of a 900° F braze alloy for Bsc/Al face skin-Ti honeycomb structural sandwich and investigation of the effects of thermal incompatibility because of relatively large differences in the thermal expansion of the aluminum matrix and boron reinforcement. Application of Bsc/Al in brazed honeycomb assemblies will be paced by the braze alloy and braze process development.

Material cost for Bsc/Al will remain high relative to boron and graphite organic matrix composite but should remain below the cost of other metal matrix composites.

Borsic/Titanium (Bsc/Ti) is lagging about 5 years in development relative to Bsc/Al (ref. 3-4). Titanium 1310 used in this combination is a beta-type alloy, composed of Ti 13V-10Mo-5Zr-2.5Al which is compatible with Borsic fiber in processing. This composite is more suitable for a Mach 3.2 SST because of its 800° F temperature capability but could be used in specialized areas of a Mach 2.7 SST.

Graphite/Aluminum composites are available in experimental quantities alone and in combination with titanium sheet. The potential for this material in airframe application is not predictable at the present time. However, successful developmental work and favorable environmental stability could make it a reasonable candidate for wing and body structure.

Other metal matrix composites, such as tungsten reinforced nickel, are not recommended for airframe primary structure in a Mach 2.7 SST. These materials are being developed for 1,700° F application in airplane engines and other very high temperature structures.

It appears that at least one system of Gr/PI and B/PI may be available for application to primary structure in a Mach 2.7 airplane by about 1980. Basic allowable properties are listed in reference 3-3 for initial properties. Design values which should be used, however, are approximately 50% of initial values to compensate for expected losses of strength due to extended 450° F exposure. All exterior areas constructed of graphite or boron require lightning protection in the form of fine aluminum wire mesh bonded to the outer surface and extended to surrounding metal structure.

Polyimide or other high temperature stable adhesives, are expected to be ready for necessary bonding of Gr/PI and B/PI structures for all composite-to-metal joint areas and for reinforcement of metal sections. Special design precautions are required to prevent undue stresses caused by differences in thermal expansion in structures combining metal and composite parts.

Development of polyphenylquinoxaline (PPQ) could offer increased thermal stability over polyimide (ref. 3-5). The work is still in an early stage but this material could be ready for primary structural applications in an SST by about 1980 using boron and/or graphite reinforcement.

FUEL TANK SEALANT

The fuel tank sealant which is recommended for use in a Mach 2.7 SST in 1980 is DC 94-530, a low modulus modification of DC 94-529. It will soon be tested in the RF-12A to obtain service experience in supersonic flight conditions. Other candidate materials include: a hybrid fluorocarbon-fluorosilicone being developed by Dow Corning under contract to AFML. The material now in test is admittedly far from optimum and other variations are being investigated. The present formulation does not have the desired flexibility at low temperatures.

Horizons Research is developing a fluorinated phosphazene. Some progress has been made in increasing the thermal stability but it is doubtful that it will ever have the necessary temperature resistance.

Other possibilities are a cyano-siloxane from Products Research and Chemical Corporation, polyfluoro-alkylene oxide from Pennisular Chemical Research, a silicone phosphoxide and diphenyl diphenyloxide cyano ethyl silicone from General Electric, polyimide from TRW Systems (TRW), and perfluorovinyl ether linked with tetrafluoro ethylene from du Pont.

Evolution of a fuel tank sealant from a polymer is a long and time consuming task. For instance, the basic polymer for DC 94-529 already existed at the time a search for a suitable SST fuel tank sealant was started in 1963. Modifications to the polymer have been continuous since 1963, after deficiencies in the material were noted during environmental testing. To have a sealant proven superior to DC 94-529 by 1980 requires extensive testing that should have begun early in 1974.

FUEL TANK INSULATION

The quality of fuel tank insulation in 1980 is not expected to be superior to current materials unless extensive research and development are undertaken. The same general comments discussed for 1975 technology insulation concepts also apply to recommendations for 1980.

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- 3-2 J. B. Kelley and R. R. June, *Residual Stress Alleviation of Aircraft Metal Structures Reinforced With Filamentary Composites*, NASA CR-112207, January 1973.
- 3-3 "Advanced Composites Design Guide (Volume I)," Los Angeles Division of North American Rockwell Corporation, Contract No. F33615-71-C-1362, November 1971.
- 3-4 C. K. Schmitz and A. G. Metcalfe, *Evaluation of Compatible Titanium Alloys in Boron Filament Composites*, AFML-TR-73-1, Solar Division of International Harvester Company, February 1973.
- 3-5 P. M. Hergenrother, *Exploratory Development Leading to Improved Phenylquinoxaline Polymers*, NASA CR-121109, The Boeing Company, Seattle, Washington, January 1973.

Candidate materials	Availability	Environmental effects	Production capability	Cost	Specifications	Selected
Metals						
Titanium						
Ti-6Al-4V	Yes	Yes	Yes	Yes	Yes	Yes
Ti-8Al-1V-1Mo	Yes	No				No
Ti-6Al-6V-2Sn	Yes	Yes	Yes	Yes	Yes	Yes
Steel						
Ph 15-7 Mo	Yes	Yes	Yes	Yes	Yes	Yes
15-5 Ph	Yes	Yes	Yes	Yes	Yes	Yes
Custom 45J	Yes	Yes	Yes	Yes	Yes	Yes
Metal sandwich						
Al brazed Ti	Yes	Yes	Yes		Yes	Yes
Stressskin	Yes	Yes	No	—	—	No
Ag brazed steel	Yes	Yes	Yes	No	—	No
Composites						
Bsc/Al	Yes	Yes	No	—	—	No
B/Pi	Yes	—	—	—	—	No
Gr/Pi	Yes	—	—	—	—	No
B/Ep	Yes	No	—	—	—	No
Gr/Ep	Yes	No	—	—	—	No
Sealants						
DC-94-529	Yes	Yes	Yes	Yes	Yes	Yes
DC-94-530	Yes	No	—	—	—	No
Insulations						
Fluorosilicone	Yes	—	—	—	—	Yes
Ti foil envelope	Yes	—	—	—	—	Yes

FIGURE 3-1.—MATERIALS SELECTION PROCESS—1975 TECHNOLOGY

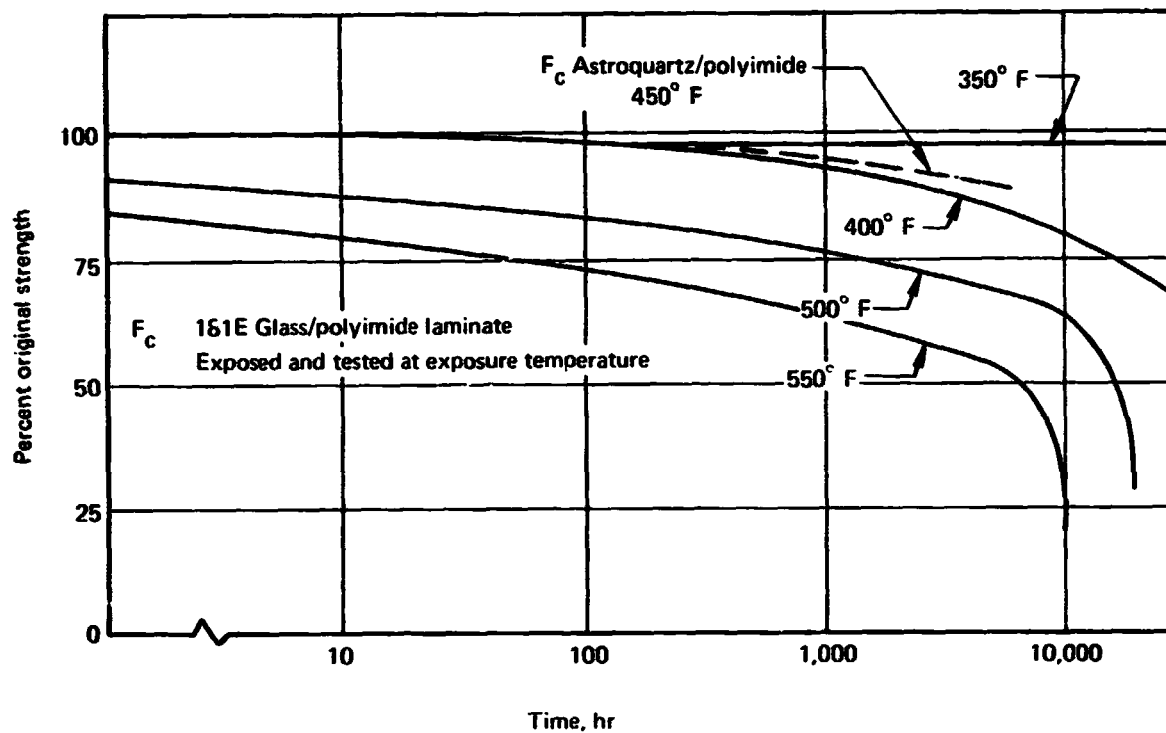


FIGURE 3-2.—TIME-TEMPERATURE EFFECTS ON POLYIMIDE COMPOSITE

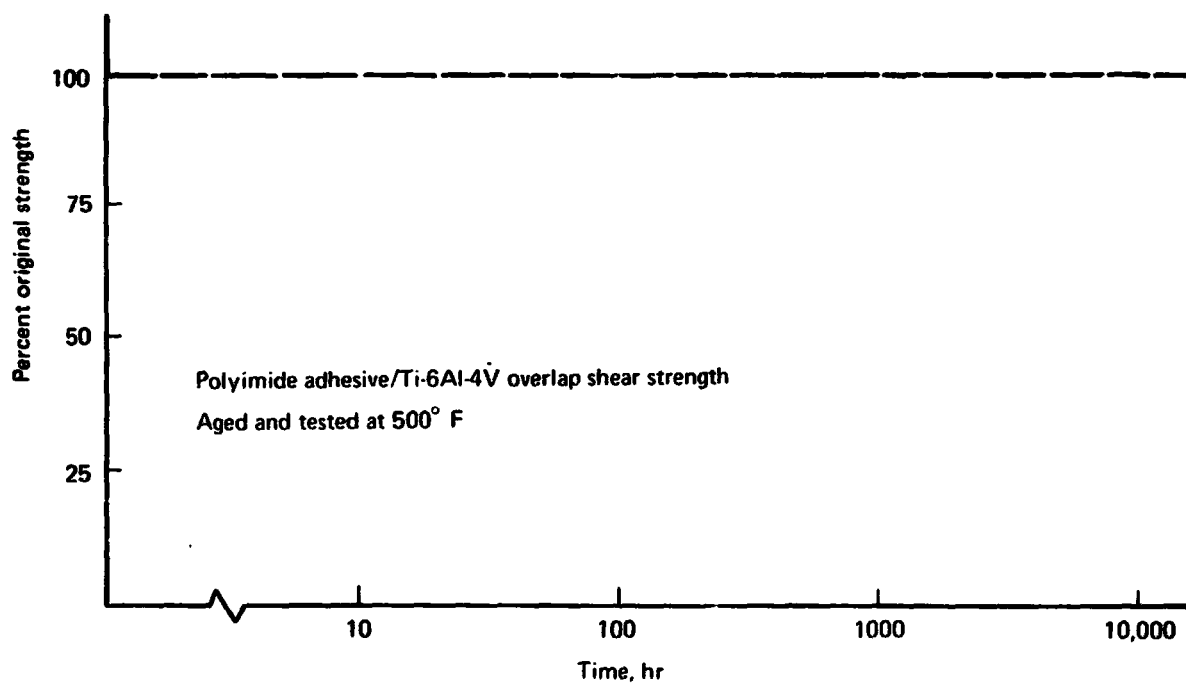


FIGURE 3-3.—TIME-TEMPERATURE EFFECTS ON POLYIMIDE ADHESIVE

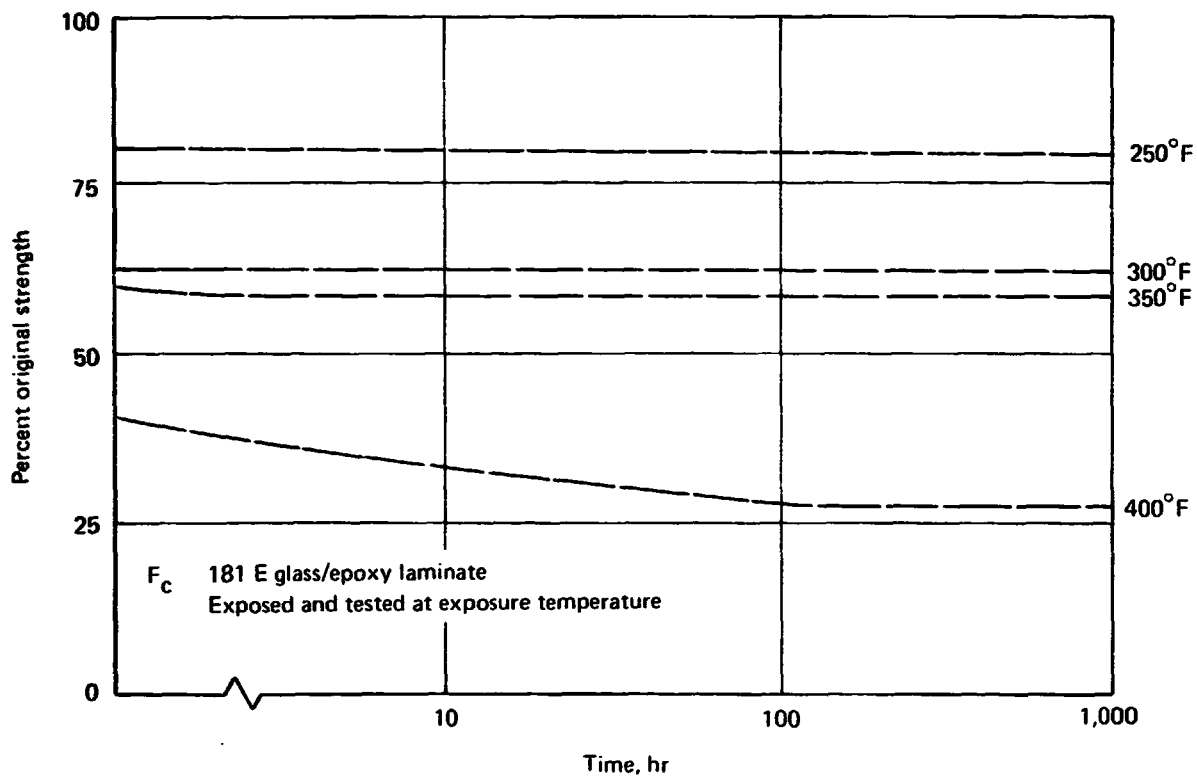


FIGURE 3-4.—TIME-TEMPERATURE EFFECTS ON EPOXY COMPOSITE

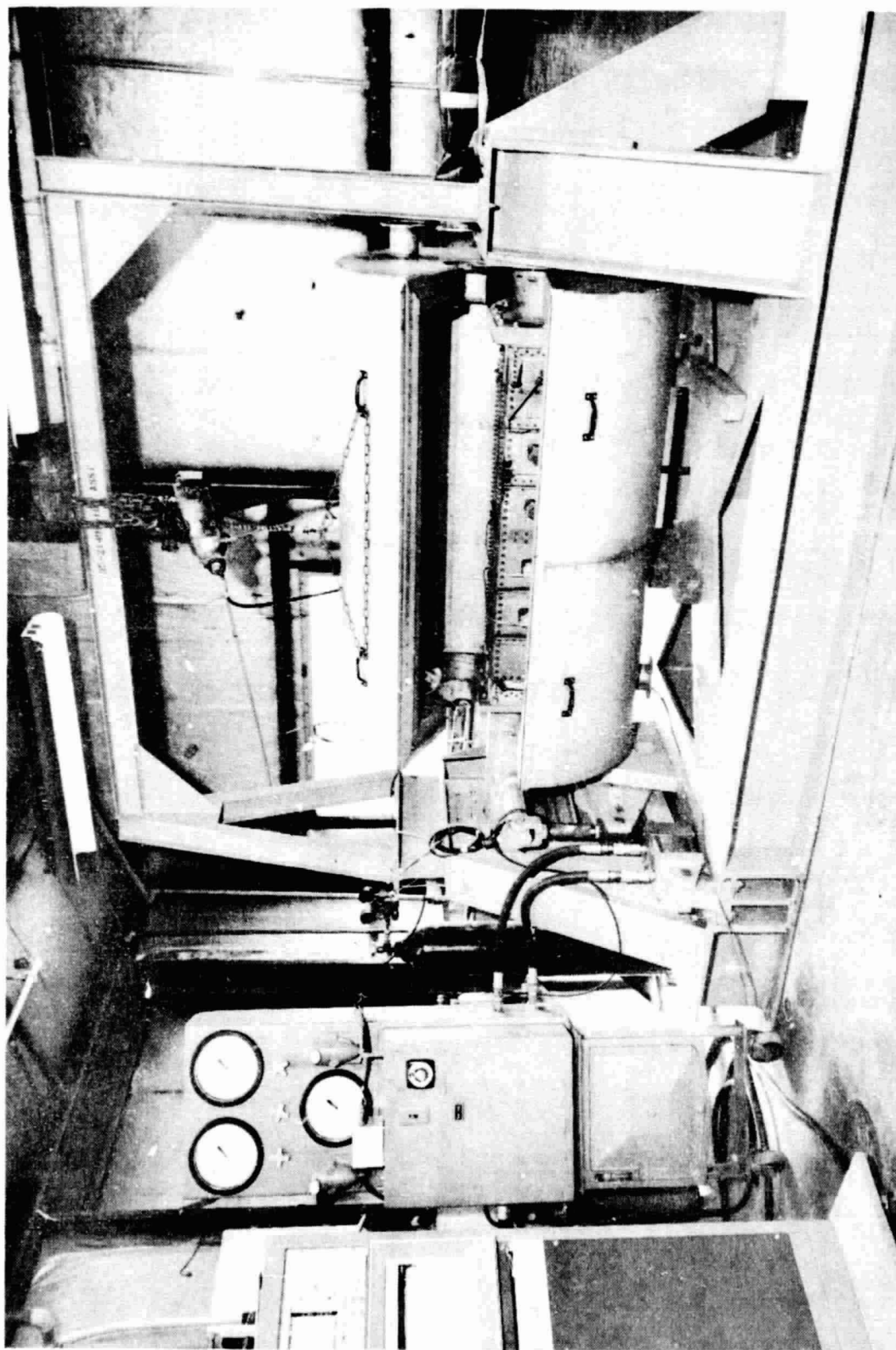


FIGURE 3-5.—FUEL TANK SEALANT TEST

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OF POOR QUALITY

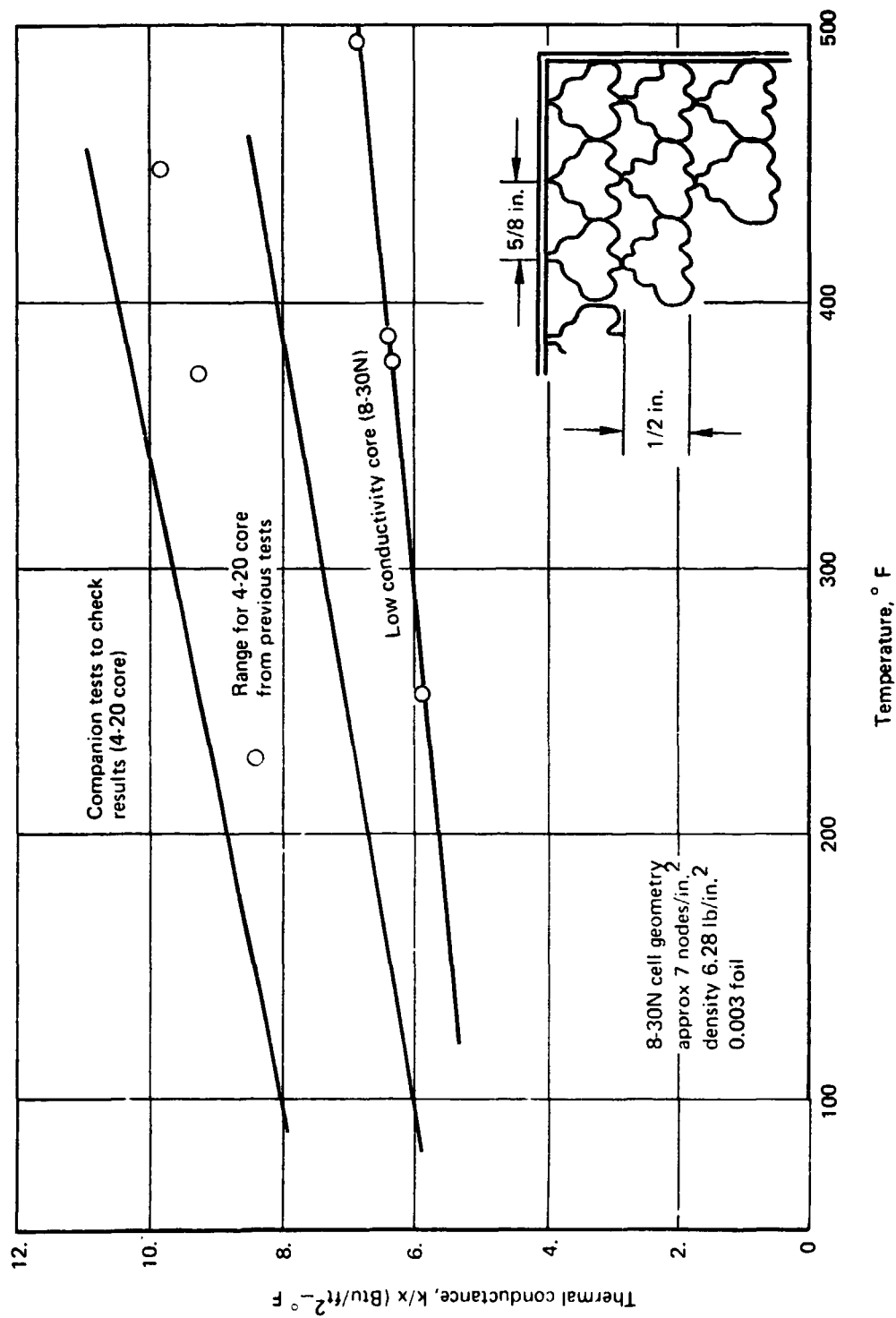


FIGURE 3-6.—CONDUCTIVITY OF CORE

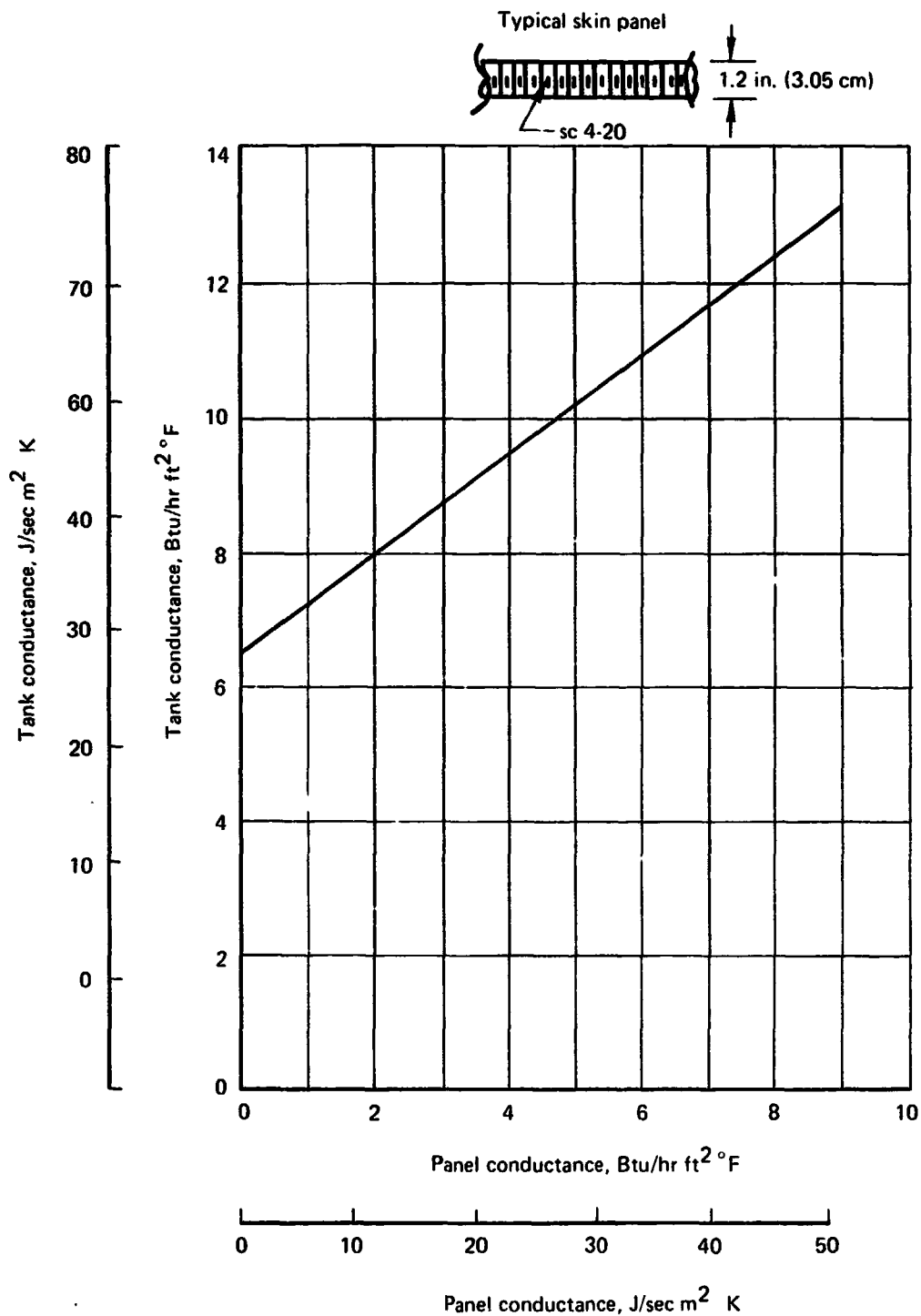


FIGURE 3-7.—PANEL/TANK CONDUCTANCE CHARACTERISTICS

Material	Relative conductivity
Titanium 6Al-4V	1.0
Magnesia	0.35
Pyrolytic graphite	0.26
Pyroceram	0.23
Silica	0.19
Zirconia	0.12
Mica	0.06
Reinforced silicone	0.05
Polytetrafluoroethylene (PTFE)	0.03
Thoria	≥ 0

FIGURE 3-8.—POTENTIAL INSULATING MATERIALS

Candidate materials	Availability	Environmental effects	Production capability	Cost	Specifications	Selected for arrow wing SST
Metals						
Titanium						
Strength design						
Ti-17	Yes	Yes	—	—	—	Yes
Ti-6-2-2-2-2	Yes	Yes	—	—	—	Yes
Beta C	Yes	Yes	—	—	—	Yes
Ti-6-2-46	Yes	Yes	—	—	—	Yes
Toughness design						
Ti-6-4-AST 1, AST 2, AST 3, AST 4	Yes	Yes	Yes	Yes	Yes	Yes
Steels						
300 M	Yes	Yes	Yes	Yes	Yes	Yes
15-5 Ph	Yes	Yes	Yes	Yes	Yes	Yes
Custom 455	Yes	Yes	Yes	Yes	Yes	Yes
MP 35 N	Yes	—	Yes	—	—	Yes
Composites						
Bsc/Al	Yes	Yes	—	—	—	Yes
Gr/Al	—	—	—	—	—	—
Bsc/Ti	—	Yes	—	—	—	—
Gr/Pi	Yes	Yes	Yes	Yes	—	Yes
B/Pi	Yes	Yes	Yes	Yes	—	Yes
B/PPQ	Yes	Yes	Yes	Yes	—	Yes
GR/PPQ	Yes	Yes	Yes	Yes	—	Yes
Fuel tank sealant						
DC-94-530	Yes	—	Yes	Yes	—	Yes
Fuel tank insulation						
Fluorosilicone	Yes	—	Yes	Yes	—	Yes
Ti foil envelope	Yes	—	—	—	—	Yes

FIGURE 3-9.—MATERIALS SELECTION—1980+ TECHNOLOGY

BORSIC/ALUMINUM-[0]

4.1 Mil Borsic $V_f = 0.50$

Design strengths	Longitudinal tensile ultimate	ksi	F_L^{tu}	140.0*
	Transverse tensile ultimate	ksi	F_T^{tu}	14.8
	Longitudinal compression ultimate	ksi	F_L^{cu}	200.0
	Transverse compression ultimate	ksi	F_T^{cu}	32.2
	In-plane shear ultimate	ksi	F_{LT}^{su}	14.5
	Interlaminar shear ultimate	ksi	F_{isu}	14.7
	Ultimate longitudinal strain	$\mu\text{in./in.}$	ϵ_L^{tu}	5,000
	Ultimate transverse strain	$\mu\text{in./in.}$	ϵ_T^{tu}	6,000
Elastic properties	Longitudinal tension modulus	msi	E_L^t	32.0
	Transverse tension modulus	msi	E_T^t	22.0
	Longitudinal compression modulus	msi	E_L^c	30.0
	Transverse compression modulus	msi	E_T^c	19.0
	In-plane shear modulus	msi	G_{LT}	8.3
	Longitudinal Poisson's ratio		ν_{LT}	0.26
	Transverse Poisson's ratio		ν_{TL}	0.16
Physical constants	Density	lb/in.^3	ρ	0.098
	Longitudinal coefficient of thermal expansion	$\mu\text{in./in.}/^\circ\text{F}$	α_L	3.2
	Transverse coefficient of thermal expansion	$\mu\text{in./in.}/^\circ\text{F}$	α_T	10.6

* $F_L^{tu} = 155$ ksi indicated for 5.7 mil borsic/aluminum

FIGURE 3-10.—BORSIC ALUMINUM ALLOWABLES

SECTION 4

STRUCTURAL CONCEPTS EVALUATION

by

H. M. Tomlinson
F. D. Flood
V. D. Bess

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Symbols and Abbreviations

A_{frame}	Frame area
A_s, A_{st}	Stiffener area
a	Length of unloaded edge of a compression panel
b	Width of a cross section or length of loaded edge of compression panel, stringer spacing
c	Fixity coefficient for columns, distance from neutral axis to extreme fiber, or thickness of sandwich core
D	Diameter, flexural rigidity of plates $(Et^3)/(12)(1-\mu^2)$ or flexural rigidity per inch of panel edge
d	Stiffener spacing, frame spacing, hole spacing, depth or height, or distance between centroids of sandwich faces
E	Modulus of elasticity in tension, average ratio of stress to strain for stress below proportional limit
E_c	Modulus of elasticity in compression, average ratio of stress to strain below proportional limit
E_{cc}	Compression modulus of elasticity of the core material in the T-direction (thickness) of the core
E_r	Reduced modulus of elasticity compression
E_s	Secant modulus
E_t	Tangent modulus
F_b	Modulus of rupture in bending
F_c, F_{ccr}	Allowable compressive stress
F_{cc}	Allowable crushing or crippling stress (upper limit of column stress for local failure)
F_{cr}	Critical compressive instability stress
F_s	Allowable shearing stress

Symbols (Continued)

F_{sall}	Allowable gross area web shear stress without curvature correction
$(F_s)^*_{all}$	Basic allowable curved web shear stress uncorrected for ring and stringer area
F_{scr}	Critical shear stress for buckling of rectangular panels
F_{scrc}	Curved panel shear buckling stress
F_{scrf}	Equivalent flat panel shear buckling stress of curved panel
F_{scrs}	Incremental shear buckling stress of curved panel resulting from curvature
F_{sL}	Allowable shear stress in the LT-plane of the core
F_{su}	Ultimate stress in pure shear (this value represents the average shearing stress over the cross section).
F_w	Allowable compressive face wrinkling stress
F_{sw}	Allowable shear stress in the WT-plane of the core
F_{ws}	Allowable shear face wrinkling stress
F_{tu}	Ultimate tensile stress (from tests of standard specimens)
f_c'	Stringer compressive stress from diagonal tension
f_c''	Equivalent compressive stress resulting from the radial load components from the diagonal tension of a curved panel
f_{eff}	Modified maximum applied stress to account for stress variation along the frame
f_s	Applied shear stress
f_t	Applied tensile stress
G	Modulus of rigidity (shear)
G_L	Modulus of rigidity for the core in the LT-plane of the core

Symbols (Continued)

G_r	Reduced modulus of rigidity
G_s	Secant modulus of rigidity
G_t	Tangent modulus of rigidity ($dF/d\gamma$) for the face material
G_{xz}	Core modulus of rigidity in the xz -plane
G_w	Modulus of rigidity for the core in the WT-plane of the core
G_{yz}	Core modulus of rigidity in the yz -plane
g	Corrugated core flat length where attached to faces
g_f	Term distinguishing stability differences of segments with one edge free and segments with no edges free
h	Height or depth, especially the distance between centroids of chords of beams and trusses, total thickness of sandwich, or distance between centroids of beam chords
h_c	Height of core in corrugated core sandwich
h_r	Distance between centroids of web-to-chord rivet patterns (effective "web" depth)
I	Moment of inertia or effective moment of inertia of shell
I_{str}	Moment of inertia of stringer
J_{st}	Torsion constant of stringer
K	Buckling coefficient for an isotropic plate under edgewise compression or diagonal tension factor
K_c	Core shear parameter for a sandwich panel under edgewise compression
K_{cs}	Average core modulus of rigidity
K_{NA}	Net area efficiency factor
K_s	Buckling constant for an isotropic plate under edgewise shear

Symbols (Continued)

K_w	Face wrinkling coefficient for a sandwich panel under edgewise compression
L	Length, circumferential distance along compression cap between points of lateral support, or ribbon direction of core
l	Effective length to be used in lateral stability equation, shear web stiffener spacing
l_c	Corrugated core flat diagonal length
l_f	Corrugated core pitch
M	Applied moment or couple, usually a bending moment
M_i	Moment per inch of edge length
M_r	Allowable moment per inch of edge length
M.S.	Margin of safety
N	Load per inch of edge length, applied at the neutral axis of the sandwich section
N_{cr}	Allowable compressive load intensity
N_t	Applied ultimate tensile load intensity
n	Cycles applied, exponent, the shape parameter for the stress-strain curve, or number of individual measurements or paired measurements
p	Applied pressure
p_w	Allowable lateral pressure on web
p_s	Allowable lateral pressure on stiffener
psi	Pounds per square inch
q	Shear flow
R	Stress ratio, radius of curved panel, or effective radius of fuselage
S	Shear force; buckling parameter for corrugated or truss sandwich panels; or core shear parameter used to determine buckling coefficients, K_{cr}

Symbols (Concluded)

s	Nominal core cell size, developed length of circular arc
t	Skin thickness or face thickness (subscripts 1 and 2 denote unequal face thicknesses)
t_c	Thickness of core in corrugated core sandwich
t_f	Thickness of face sheet in corrugated core sandwich
t_s	Thickness of attached stiffener flange on shear web
w	Distributed loading
$2w$	Effective width of buckled skin
Δ	Curved web correction factor
ΔC_s	Coefficient related to increase in shear buckling stress of curved panels due to internal pressure
ϵ_{tu}	Unit tensile strain corresponding to F_{tu}
μ	Poisson's ratio
ρ	Radius of gyration

SECTION 4

STRUCTURAL CONCEPTS EVALUATION

INTRODUCTION

The structural concepts defined in Section 2 have been evaluated and the most satisfactory concepts selected for use on the various areas of the airplane. It is, of course, impossible to consider the design of the complete airplane with both limited time and budget. Therefore, three specific locations on the wing and ten locations on the fuselage were used as evaluation points and the applicable concepts studied at each point.

Each applicable concept was compared to the others at these points, considering factors such as weight, manufacturing complexity, stiffness, maintainability, fatigue, fail safety, thermal conductance, and material cost. The following paragraphs describe in greater detail the evaluation procedure and final selections.

DESIGN LOADS AND CRITERIA

The design loads and criteria used for preliminary sizing and structural concept evaluation are the flight and ground loads determined for the 969-336C airplane and summarized in reference 4-1. The flight loads contained therein represent the critical, symmetric maneuver conditions at both positive and negative load factors. These loads were obtained from an aeroelastic analysis in which the airplane's aerodynamic characteristics were computed on 93 panels using a planar approximation of the linearized potential flow theory. The elastic nature of the structure was accounted for through use of a flexibility matrix generated from a finite element model of the airplane's structure.

Landing impact loads are based on a dynamic analysis which included the first six symmetric elastic modes, airplane vertical translation and pitch. The airplane is trimmed with the aerodynamic lift equal to the gross weight at impact.

The loads in the taxi condition are based on a static analysis in which the airplane center of gravity is assumed to experience a limit load factor of 2.0 and the inertia loads are balanced by vertical ground reactions at the wheels.

The fuselage panel design loads used to evaluate the structural concepts are listed in figure 2-3 with the panel locations defined in figure 2-7. These fuselage loads are derived from the shear and bending moment curves presented in figures 4-1 and 4-2 respectively. The ultimate design loads and locations used to evaluate the structural concepts for wing panels and spars are shown in figure 2-1.

STRUCTURAL SIZING

This section presents the analytical procedures used to size the various structural concepts selected for evaluation. Concepts for wing panels, wing spars and fuselage panels were sized to design strength requirements. The control points at which the concepts were sized were selected to represent diverse design conditions as well as an extensive range of load magnitude. The design conditions (loads and thermal environment) used for the concept sizing were taken from the SCAT-15F study (ref. 4-1). The geometry, such as spar spacing, frame spacing and spar depth, is that of the model 969-512B at the particular control point.

The concepts, sized by these procedures, were then evaluated for weight, shear stiffness, fatigue and fail safe capabilities. The resultant values for the various concepts considered, supplemented by values for the additional rating attributes, formed the basis for the structural concept selection reported below.

WING PANEL CONCEPTS

The analysis procedures used to size the various wing panel concepts are presented in this section. Control points were selected on the wing planform for evaluation of concepts for three upper and three lower surface panels. These points represent a spectrum of wing panel loading ranging from the lightly-loaded strake region forward of the wheel well to the highly-loaded main box aft of the wheel well. The critical design conditions, loads and thermal environment are shown in figure 2-1. For lower surface concepts, 40% bending reversal was assumed to obtain compressive design loads. Instability failure under biaxial compression and shear is discussed for each concept in the applicable paragraph below. For biaxial tension and shear, the panel was checked for maximum principal tension and maximum shear stress. The tension allowable for all panels (multi-load path) is 0.9 of the "B" value for F_{Tu} , while the shear allowable is 0.9 of the "B" value for F_{Su} . The 0.9 factor derives from a 10% allowance for the combined effects of area-out and section net area efficiency. Thus, until these combined effects exceed 10%, only the gross area stresses need be considered. This is consistent with design practice for wing panel sizing. Requiring each wing panel to be sized for net area stresses for such a low proportion of area-out would disproportionately increase the effort required relative to the accrued benefits.

Design details for the spanwise splices are in accordance with design practices for integral fuel tank panels. Stiffener-to-panel attachments were designed to provide load transfer capability for fail safe requirements.

With the limited loads data available, the fatigue evaluation of the strength-sized panels was based on ground-air-ground (GAG) cycle stresses with fatigue reliability factors, fatigue detail ratings and GAG damage ratios estimated from the National SST Program results.

Skin-Stringer

The analysis procedure used to size the skin-stringer panels is presented in this section. Two geometrically distinct but analytically similar concepts are considered—riveted skin-stringer and integral skin-stringer. Both concepts have primary stringers oriented in the spanwise direction. The integral skin-stringer panels have secondary, chordwise unflanged stringers at the same pitch as the primary stringers at wing control points having high chordwise compression loads. Figure 4-3 illustrates these two skin-stringer concepts and defines the geometric notation used below.

Skin buckling is considered first. Since no other structure exists to carry the chordwise compression loads, the skin must not buckle prior to ultimate load when subjected to combined biaxial compression and shear.

The allowable spanwise compressive stress for skin buckling is given by

$$\frac{F_{cr}}{E_r} = K \left(\frac{t}{b} \right)^2 \quad (4-1)$$

where K is the classical plate buckling coefficient. Since the panel aspect ratio is large (or with secondary stiffeners is unity) and the panel edges are assumed to be simply supported, K is taken as the asymptotic value 3.62. From a graph of F vs. F/E_r determine F_{cr} . This graph is based on the reduced modulus $E_r = \sqrt{EE_t}$. Finally, the ratio of applied to allowable stress is determined.

The allowable chordwise compressive stress is given by

$$\frac{F_{cr}}{E_r} = K \left(\frac{t}{b} \right)^2 \quad (4-2)$$

where K is taken as

1. 0.904 in the absence of secondary chordwise stiffeners (this corresponds to wide column behavior of length b).
2. 3.62 in the presence of secondary chordwise stiffeners which have a pitch b equal to that of the primary stiffeners.

As above, F_{cr} is determined from a graph of F vs. F/E_r ; then, the ratio of applied to allowable stress is determined.

The allowable shear stress for skin buckling is given by

$$\frac{F_{s,cr}}{G_s} = K \left(\frac{t}{b} \right)^2 \quad (4-3)$$

where K_s is taken as 12.56 for the large aspect ratio panel with simply supported edges. From a graph of F_s vs. F_s/G_s determine F_{scr} . This graph is based on the secant shear modulus. Then, determine the ratio of applied to allowable shear stress.

To predict the skin buckling margin of safety for combined biaxial compression and shear, the method of stress ratios presented in reference 4-2 for a panel aspect ratio of one (four) for a section having (no) secondary stiffeners is used.

The next step in the analysis is to determine the critical spanwise stress for the skin-stringer (primary) combination. This is outlined in the following steps:

1. Determine the stringer crippling stress F_{cc} . For each flange element, determine its length-to-thickness ratio b/t . Divide this ratio by the term g_f which distinguishes stability differences of segments with one edge free and segments with no edges free. For Ti-6Al-4V Condition 1, $g_f = 1.0(2.3)$ for an element having one (no) edge free. With this value of b/g_ft , enter an empirical crippling curve for Ti-6Al-4V to determine the i^{th} flange crippling stress F_{cci} . For a stringer having n flanges, the stringer crippling stress is given by

$$F_{cc} = \frac{\sum_{i=1}^n (b_i t_i F_{cci})}{\sum_{i=1}^n (b_i t_i)} \quad (4-4)$$

2. For a section composed of a primary stringer and its associated skin (one stringer pitch wide), determine the section radius of gyration ρ .
3. Determine the slenderness ratio L/ρ of the section based on the rib or former spacing L assuming the rib or former provides simple support to the section.
4. With the calculated values for the crippling stress F_{cc} and the slenderness ratio L/ρ , calculate the Johnson-Euler column allowable

$$F_c = K_L F_{cc} \left[1 - \frac{K_L F_{cc} (L/\rho)^2}{4\pi^2 E} \right] \quad (4-5)$$

where $K_L = 1 + 4F_{cc}/E$.

5. Calculate the Euler-Engesser column allowable

$$F_c = \frac{\pi^2 E_t}{(L/\rho)^2} \quad (4-6)$$

6. The lower of these two column allowables, or the spanwise skin buckling allowable, is taken as the critical spanwise stress for the skin-stringer (primary) combination.

For sections having secondary chordwise stringers, determine the critical chordwise stress for the skin-stringer (secondary) combination. The procedure to predict this allowable is analogous to that used for the skin-stringer (primary) combination, except the length is taken equal to the primary stringer spacing. The critical chordwise stress is again taken equal to the lower of the two column allowables or the chordwise skin buckling allowable.

Additionally, the stringer moment of inertia must be at least the value given by the following equation to provide stability under the shear loading.

$$I = \frac{d}{t} \left[\frac{h_r^2}{30.7} \frac{t}{G_r} \frac{f_s}{G_r} \right]^{4/3} \quad (4-7)$$

This equation is for a skin and stringer of common material. The parameters are

d - stringer spacing

t - skin gage

h_r - effective "web" depth

f_s - applied shear stress

G_r - reduced shear modulus $\sqrt{G_t G_l}$

For a primary (spanwise) stringer, the following geometry pertains

d - primary stringer spacing

h_r - rib or former spacing

For a secondary (chordwise) stringer, the following geometry applies

d - secondary stiffener spacing

h_r - primary stringer spacing.

Integrally Machined Waffle

The analysis procedure used to size the integrally machined 0°-90° waffle panels is presented in this section. The following assumptions are made:

1. Optimum design occurs when all modes of instability occur simultaneously.
2. The rib-to-skin fillet (see fig. 4-4) is ignored.
3. Shear loads are carried in the skin only.
4. Simply supported edges are assumed at the panel boundaries and at the rib-skin intersections.
5. Axial loads are carried in both the skin and the stiffeners (ribs) parallel to the axial load.

The geometry and nomenclature are shown in figure 4-4. The waffle has square pockets and identical ribs in both directions.

For local rib instability under compressive load, the rib is considered to be a long plate with loaded edges and one unloaded edge simply supported; while the other unloaded edge is free. The critical load intensities in the x- and y-directions are given by

$$N_{x_{cr}}^{ST} = 0.388 E_s \bar{t} (t_w b_w)^2 \quad (4-8)$$

$$N_{y_{cr}}^{ST} = 0.388 E_s \bar{t} (t_w b_w)^2 \quad (4-9)$$

The 0.388 factor is the asymptotic value for an infinitely long plate with the noted boundary conditions.

Local instability of the skin for each of the applied loads is given by

$$N_{x_{cr}}^{SK} = 3.62 E_s \bar{t} (t_s b_s)^2 \quad (4-10)$$

$$N_{y_{cr}}^{SK} = 3.62 E_s \bar{t} (t_s b_s)^2 \quad (4-11)$$

$$N_{xy_{cr}}^{SK} = 8.15 E_s t_s (t_s b_s)^2 \quad (4-12)$$

The factors 3.62 and 8.15 for the compressive and shear instability derive from the plate aspect ratio of unity with the assumed simply supported edges. Denoting the applied load intensities as N_x , N_y and N_{xy}

define β as

$$\beta = 1 \text{ for } N_y/N_x \leq 1 \quad (4-13)$$

or

$$\beta = N_x/N_y \text{ for } N_y/N_x > 1 \quad (4-14)$$

The ratios of applied to allowable stress are given by

$$R_x = N_x/N_{xcr} \quad (4-15)$$

$$R_y = N_y/N_{ycr} \quad (4-16)$$

$$R_{xy} = N_{xy}/N_{xycr} \quad (4-17)$$

Noting by assumption 3. that a rib is loaded only by compressive loads parallel to that rib gives

$$N_x = N_{xcr}^{ST} \text{ for } N_y/N_x \leq 1 \quad (4-18)$$

or

$$N_y = N_{ycr}^{ST} \text{ for } N_y/N_x \geq 1 \quad (4-19)$$

and substituting in the above stress ratios with the proviso that the rib and skin local instabilities occur simultaneously gives

$$R_x = N_x/N_{xcr}^{SK} = \beta N_{xcr}^{ST}/N_{xcr}^{SK} = 0.388 \beta r_t^2 / 3.62 r_b^2 \quad (4-20)$$

$$R_y = N_y/N_{ycr}^{SK} = (N_y/N_x) \beta N_{xcr}^{ST}/N_{ycr}^{SK} = 0.388 (N_y/N_x) \beta r_t^2 / 3.62 r_b^2 \quad (4-21)$$

$$R_{xy} = \frac{N_{xy}}{N_{xy,cr}^{SK}} = (N_{xy} - N_x) \beta K A \frac{N_{xy,cr}^{ST}}{N_{xy,cr}^{SK}} = 0.388 (N_{xy} - N_x) \beta K A r_t^2 / 8.15 r_b^2 \quad (4-22)$$

where

$$r_t = t_w / t_s$$

$$r_b = b_w / b_s$$

$$KA = t_s^3 / t_w = 1 + r_b r_t$$

For the skin local instability mode of failure, the critical combination of biaxial compression and shear loads is given by

$$R_x + R_y + R_{xy}^2 = 1 \quad (4-23)$$

General instability of the panel for transverse (Y) compression, longitudinal (X) compression, or shear acting singly is given by

$$\frac{N_{xy,cr}^{GI}}{N_{xy,cr}^{SK}} = \pi^2 D_{ST} / L^2 \quad (\text{wide column formula}) \quad (4-24)$$

$$\frac{N_{xy,cr}^{GI}}{N_{xy,cr}^{SK}} = 2\pi^2 D_{ST} / L^2 \quad (\text{ref. 4-3 and ref. 4-4, Part VII}) \quad (4-25)$$

$$\frac{N_{xy,cr}^{GI}}{N_{xy,cr}^{SK}} = 4\pi^2 D_{ST} / L^2 \quad (\text{ref. 4-3, Part VII}) \quad (4-26)$$

where

$$D_{ST} = D_s K_1 (b_w t_w)^2$$

$$D_s = E_s t_s^3 / [12 (1 - \mu^2)]$$

and from the Appendix of reference 4-5

$$K_1 = 1.095 r_b r_t^3 \left[1 + \frac{3}{KA} \right]$$

The coefficients 2 and 4 in the equation for N_{xcr}^{GI} and N_{xycr}^{GI} , respectively, were assumed for preliminary design purposes and may underpredict the respective allowables.

Substituting for the geometric parameters in the equation for N_{ycr}^{GI} and noting that $N_x = \beta N_{xcr}^{ST}$ gives

$$N_{ycr}^{GI}(EL) = \lambda (\bar{t} L)^4 \quad (4-27)$$

where

$$\lambda = \left\{ 0.388 \pi^2 K_1 [12 (1 - \mu^2) (KA)^3] \right\} (EL N_x) \beta$$

Similar substitutions into the equations for N_{xcr}^{GI} and N_{xycr}^{GI} yields, respectively,

$$N_{xcr}^{GI}(EL) = 2\lambda (\bar{t} L)^4 \quad (4-28)$$

and

$$N_{xycr}^{GI}(EL) = 4\lambda (\bar{t} L)^4 \quad (4-29)$$

The interaction for panel general instability under combined biaxial compression and shear is given in terms of stress ratios as

$$\left[\frac{R_x}{1 - R_{xy}^2} \right]^4 + \frac{R_y}{1 - R_{xy}^2} \leq 1 \quad (4-30)$$

This assumes a panel aspect ratio of at least 3.5.

The analytical procedure set forth in this section has been automated on the CDC 6600 computer. The automated procedure consists of a solution of the skin local instability interaction formula (as constrained by rib instability) which gives

$$r_b = f_1(r_t) \quad (4-31)$$

and the panel general instability interaction formula (as constrained by local instability), giving

$$N_{xy}(EL) = f_2(\bar{t}/L) \quad (4-32)$$

For a given set of loads and panel width, a value is assumed for r_t . The program then iteratively solves for the detail panel geometry which has the minimum value of $\bar{t}/L$ (i.e., least weight). Design charts are available for a wide range of N_{xy}/N_x and N_y/N_x values.

Cellular Core Sandwich

The equations used to size the cellular core sandwich panels are presented in this section. The general instability and local instability modes of failure are considered.

The allowable spanwise compressive stress (general instability) for a sandwich with isotropic faces of unequal thickness t_1 and t_2 is given by

$$F_c = \frac{\left(3E_r \frac{2hK_c}{t_1 + t_2}\right)}{\left[\left(\frac{b}{d\sqrt{K}}\right) \frac{t_1 + t_2}{2\sqrt{t_1 t_2}}\right]^2 \left(\frac{2hK_c}{t_1 + t_2}\right) + E_r} \quad (4-33)$$

where $E_r = 2EE_t/(E + E_t)$, and for all edges simply supported

$$K_c = \frac{1}{6} \left[G_{xz} + G_{yz} \left(\frac{a}{nb} \right)^2 \right] \quad (4-34)$$

In the formula for K_c , y is the loaded (spanwise) direction and the number of half-wave buckles n is taken equal to the panel aspect ratio a/b . Since the typical panel aspect ratio is large, the buckling coefficient K is taken as the asymptotic value 3.62. The core shear moduli are obtained by test and are typically presented as shown in figure 4-5.

The allowable chordwise compressive stress for general instability is conservatively taken as the wide-column allowable. The equations above apply except for loaded edges simply supported with remaining edges free

$$K_c = \frac{G_{yc}}{6} \text{ and } K = \frac{\pi^2}{12(1-\mu^2)} \left(\frac{b}{a}\right)^2 \quad (4-35)$$

where again, b is the length of the loaded edge and y is the loaded (chordwise) direction.

The allowable inplane shear stress (general instability) for a sandwich with isotropic faces of unequal thickness, t_1 and t_2 is given by

$$F_s = \frac{G_r}{2 \left[\frac{1}{6} \left(\frac{b}{a\sqrt{K_s}} \frac{t_1 + t_2}{2\sqrt{t_1 t_2}} \right) + \left(\frac{G_r}{2hK_{cs}} \right) \right]} \quad (4-36)$$

where

$$G_r = \frac{2GG_t}{G + G_t}, K_{cs} = \frac{1}{2} (G_{xz} + G_{yz})$$

and K_s is the shear buckling coefficient for the panel aspect ratio a/b . For the typical large panel aspect ratio, a value of 12.56 is taken for K_s .

To predict the general instability margin of safety for combined biaxial compression and shear, the method of stress ratios presented in reference 4-2 for a panel aspect ratio of four is used.

The local instability modes of failure are considered in the following paragraphs. These modes are intracell buckling, face wrinkling and shear (core) crimping.

The allowable compressive intracell buckling stress for a sandwich with isotropic faces is given by

$$F_c = \frac{\pi^2 E_t}{3(1-\mu^2)} \left(\frac{t}{s}\right)^2 \quad (4-37)$$

where s is the diameter of a circle inscribed in a core cell, and t is the thinner of the two face sheets. This is seen to be equivalent to compressive buckling of a simply supported isotropic plate when the ratio of the length of the unloaded edge is any

integer multiple of the length of the loaded edge. Note that the allowable is independent of the core orientation.

The allowable shear intracell buckling stress for a sandwich with isotropic faces is given by

$$F_s = \frac{1.45 \pi^2 G_L}{1 - \mu} \left(\frac{t}{s} \right)^2 \quad (4-38)$$

where s and t are defined as in the preceding paragraph. Again, this allowable is independent of core orientation.

To predict the intracell buckling margin of safety for combined biaxial compression and shear, the method of stress ratios (ref. 4-2) for a panel aspect ratio of unity is used.

The allowable compressive face wrinkling stress is given by

$$F_w = K_w \eta (G_L t_{cc})^{1/3} \quad (4-39)$$

where $\eta = [(3E_L + E_S)/4]^{1/3}$ and $K_w \approx 0.6$. This formula considers failure from core crushing and face separation.

The allowable compressive load intensity for core shear crimping failure is

$$N_{cr} = 0.75 \frac{d^2}{c} G_L \quad (4-40)$$

if the applied load parallels the core L-direction. For applied load parallel to the core W-direction, the appropriate core shear modulus G_W is used.

Since the margins of safety for face wrinkling and core shear crimping for the major component of compression load (spanwise) were large, the effect of combined biaxial compression and shear loads on these failure modes was not considered.

Corrugated Core Sandwich

The analysis procedure used to size the corrugated core sandwich panels is outlined below. The general instability and local instability modes of failure are considered. Figure 4-6, a cross section, defines the notation used. The core is oriented to be effective for spanwise (longitudinal) loads.

This procedure correlates well with the data (curves) presented in reference 4-1 for 30-inch spar spacing. However, the 35-inch spar spacing for the model 969-512B required calculation of new allowables for general instability.

The allowable spanwise compressive stress for general instability for a sandwich having equal-thickness isotropic faces is obtained by the following steps:

1. Calculate

$$U = \frac{Sd E_c}{1 - \mu^2} \left(\frac{t_c}{h_c} \right)^3 \quad (4-41)$$

where S is obtained from charts based on reference 4-6.

2. Calculate

$$C = \frac{\pi^2 E_c t_f d^2}{2 (1 - \mu^2) b^2} \quad (4-42)$$

3. Calculate $J = U/C$

4. Then $N_x = \dot{K}_x C$, where $\dot{K}_x = K_x \lambda_1$ and K_x and λ_1 are from charts of reference 4-7.

5. Calculate

$$\frac{F}{E_r} = \frac{N_x E_c}{2t_f + 2(g + t_c) \frac{t_c}{t_f}} \quad (4-43)$$

6. From a graph of F vs. F/E_r for Ti-6Al-4V (condition 1), determine F. This graph is based on a reduced modulus

$$E_r = \frac{2E_c E_t}{E_c + E_t} \quad (4-44)$$

7. Calculate the ratio of applied stress/allowable stress.

The allowable chordwise compressive stress for general instability is obtained by:

1. Calculate $N_y = K_y C$, with J, U, and C from above. K_y is from a chart of reference 4-7.

2. Calculate

$$\frac{F}{E_r} = \frac{N_y E_r}{2t_f} \quad (4-45)$$

3. From the graph of F vs. F/E_r for Ti-6Al-4V, obtain F .

4. Calculate the ratio of applied stress/allowable stress.

The allowable inplane shear stress for general instability is obtained by:

1. Calculate $Q = \dot{K}_s C$, with J , U and C from above, and $\dot{K}_s = K_s \lambda_2 \lambda_3$. K_s , λ_2 and λ_3 are from charts of reference 4-7.

2. Calculate

$$\frac{F_s}{G_s} = \frac{Q G}{2t_f + 2 \frac{t_c t_f}{(t_c + g)}} \quad (4-46)$$

3. From a graph of F_s vs. F_s/G_s for Ti-6Al-4V, determine F_s . This graph is based on the secant shear modulus.

4. Calculate the ratio of applied stress/allowable stress.

To predict the general instability margin of safety for combined biaxial compression and shear, the method of stress ratios of reference 4-2 for a panel aspect ratio of four is used.

The test data of reference 4-8 for biaxial compression indicate this method is only slightly conservative when the ratio of transverse to longitudinal load is small. The shear test data of reference 4-8 correlates well with this method which incorporates the correction factor of reference 4-7. Test data are not available for combined biaxial compression and shear. This method may be slightly conservative based on the correlations noted above.

The allowable spanwise compressive stress for local instability is determined as outlined below. The sandwich must be checked for core and face sheet failure.

1. Calculate the core fixity coefficient

$$\epsilon_c = \left(3 + \cos \frac{\pi g}{\ell_f} \right) \frac{\ell_c}{\ell_f} \left(\frac{t_f}{t_c} \right)^3 \quad (4-47)$$

2. Calculate

$$\frac{F}{E_r} = \frac{\pi^2 K_c}{12 (1 - \mu^2)} \left(\frac{t_c}{\ell_c} \right)^2 \quad (4-48)$$

where K_c is from the chart of reference 4-9.

3. From a graph of F vs. F/E_r for Ti-6Al-4V, determine F (for core failure).
4. Calculate the face sheet fixity coefficient

$$\epsilon_f = \left(3 + \cos \frac{\pi g}{\ell_f} \right) \frac{\ell_f}{\ell_c} \left(\frac{t_c}{t_f} \right)^3 \quad (4-49)$$

5. Calculate

$$\frac{F}{E_r} = \frac{\pi^2 K_f}{12 (1 - \mu^2)} \left(\frac{t_f}{\ell_f} \right)^2 \quad (4-50)$$

where K_f is from the chart of reference 4-9.

6. From a graph of F vs. F/E_r for Ti-6Al-4V, determine F (for face sheet failure).
7. Using the lower of these two allowable stresses, calculate the ratio of applied/allowable stress.

The allowable chordwise compressive stress for local instability is obtained by (face sheet critical):

1. Calculate

$$\frac{F}{E_r} = \frac{\pi^2 K}{12 (1 - \mu^2)} \left(\frac{t_f}{\ell_f} \right)^2 \quad (4-51)$$

where K is from the chart of reference 4-10.

2. From a graph of F vs. F/E_T for Ti-6Al-4V, determine F .
3. Calculate the ratio of applied/allowable stress.

The allowable shear stress for local instability is obtained in the following steps. Core and face sheet failure must be investigated.

1. Calculate

$$\frac{F_s}{G_s} = 3.15 K_c \left(\frac{t_c}{l_c} \right)^2 \quad (4-52)$$

where K_c is the coefficient determined for local instability under longitudinal (spanwise) load.

2. From a graph of F_s vs. F_s/G_s for Ti-6Al-4V, determine F_s (core local buckling stress).
3. Calculate

$$\frac{F_s}{G_s} = 3.15 K_f \left(\frac{t_f}{l_f} \right)^2 \quad (4-53)$$

where K_f is the coefficient for face sheet local instability under longitudinal load.

4. From a graph of F_s vs. F_s/G_s for Ti-6Al-4V, determine F_s (face sheet buckling stress).
5. Using the lower of these two allowables, calculate the ratio of applied/allowable stress.

To protect the local instability margin of safety for combined biaxial compression and shear, the method of stress ratios of reference 4-2 for a panel aspect ratio of four is used. Note that this margin of safety is applicable only for face sheet failure since the core is loaded only by spanwise load and shear. The core local instability margin of safety is obtained from the interaction equation ($R_c + R_s^2 \leq 1$) for spanwise compression and shear. The margin of safety is given by

$$M.S. = \frac{2}{R_c + \sqrt{R_c^2 + 4R_s^2}} - 1 \quad (4-54)$$

The local instability margin of safety is the lower of these two values.

Test data are available for local buckling failure of corrugated core sandwich under longitudinal, transverse, and biaxial compression loads (reference 4-8 and Boeing test data from the National SST Program). Correlation of this analysis method and these test data indicates the method is

1. conservative for longitudinal load (eight geometrically identical Ti-6Al-4V panels exhibited test stresses varying from 80 to 111 ksi-72 ksi predicted)
2. within $\pm 10\%$ for transverse load
3. generally unconservative for biaxial compression for 17- 7PH (TH 1050) panels, but slightly conservative for Ti- 6Al-4V panels.

WING SPAR CONCEPTS

The analysis procedures used to size the candidate wing spar concepts are presented in this section. Each concept was sized at each of the three wing control points previously noted. As for the wing panels, these control points provide a wide range for the spar design parameters. Since the design loads for the wing spars were not available from the SCAT-15F study documentation (ref. 4-1), the baseline structure defined therein was evaluated to determine its load-carrying capabilities. These loads were then used to design the competing wing spar concepts at each control point as outlined in the subsequent sections.

Sheet-Stiffener

The analysis procedure used to size the sheet-stiffener spars is presented in this section. Two geometrically distinct but analytically similar concepts are considered-riveted sheet-stiffener and integral sheet-stiffener. For shear loading, Wagner's criterion indicates that for all designs considered, the intermediate tension field shear webs are the most efficient design. That is $\sqrt{S}/h < 7$ where S is the shear load in pounds and h is the web depth in inches.

Lipp's method, as outlined below, was used to design and strength check the spar designs under shear loading. The allowable web shear flow q for a stiffener area (A_s) to web area (ℓt) ratio of .5 is given by

$$q = 0.37 t F_{TU} \quad (4-55)$$

where t is the web thickness and F_{TU} is the web material ultimate tension allowable. For values of $A_s/(\ell t)$ other than 0.5, the allowable shear flow is corrected by the factor k from figure 4-7.

Having established the web gage required for the applied shear loading, the stiffener requirements for the intermediate web in shear are

- The stiffener spacing ℓ should be 6 to 9 inches.

- The slenderness ratio (h/ρ) of the stiffener should not exceed 80 (ρ is the radius of gyration of the stiffener section about its centroidal axis parallel to the plane of the web.)
- The stiffener-to-skin area ratio ($A_s/\ell t$) should not be less than 0.3.
- The stiffener moment of inertia I to prevent shear general instability before web buckling is given by $I = 0.321 h t^3 (\ell/h)^{-5/3}$
- For stiffeners riveted to the web, the following constraints on the attached stiffener leg thickness t_s relative to the web thickness t should be observed:

$$\text{for } t < 0.032, t_s = 2t$$

$$0.032 \leq t \leq 0.050, t_s = 1.75t$$

$$t > 0.050, t_s = 1.5t$$

The web-to-chord attachment requirement is a function of the ratio of the applied shear stress f_s to the web shear buckling stress $F_{s_{cr}}$. The ratio of the required attachment shear flow q_c to the applied shear flow q is given below for selected ranges of $f_s/F_{s_{cr}}$.

For

$$f_s/F_{s_{cr}} < 4, q_c/q = .1993 \log (f_s/F_{s_{cr}}) + 1.0$$

$$4 \leq f_s/F_{s_{cr}} < 10, q_c/q = .1777 \log (f_s/F_{s_{cr}}) + 1.013$$

$$10 \leq f_s/F_{s_{cr}} < 30, q_c/q = .149 \log (f_s/F_{s_{cr}}) + 1.041$$

$$30 \leq f_s/F_{s_{cr}} < 100, q_c/q = .1097 \log (f_s/F_{s_{cr}}) + 1.098$$

For single stiffeners riveted to the web, the attachment requirement in terms of effective shear flow q_s is given by $q_s = 0.85q (A_s/\ell t)$. The tensile strength requirement of 0.29q is necessarily satisfied if conventional protruding head rivets are used.

After the fastener patterns to satisfy the above attachment requirements are defined, the web must be checked to verify that the web net area shear stress does not exceed the web material ultimate shear stress.

To complete the sizing, the web and stiffener must be sized for the lateral load resulting from refueling overpressurization or the 6.0g crash condition. It is noted that the stiffener is on the side of the web opposite to which the pressure is applied. Thus the web-to-stiffener load is transmitted by bearing rather than by rivet tension and the stiffener outstanding flange will be in tension. It is assumed that the web acts as a diaphragm between adjacent stiffeners in transmitting the pressure load. It is further assumed that the web as sized by the shear load requirements works in diaphragm tension to the strain ϵ_{TU} corresponding to the material ultimate tension allowable. Thus the ratio of the stiffener spacing ℓ to the developed length s of the circular arc diaphragm between adjacent stiffeners is given by

$$\frac{\ell}{s} = \frac{\ell}{\ell(1 + \epsilon_{TU})} = \frac{1}{1 + \epsilon_{TU}} \quad (4-56)$$

From the geometry of the circular arc of length s and radius R which subtends a chord of length ℓ and a central angle 2θ it follows that

$$\frac{\ell}{s} = \frac{2R \sin \theta}{2R\theta} = \frac{\sin \theta}{\theta} \quad (4-57)$$

Expanding the sine function into a Maclaurin's Series.

$$\frac{\ell}{s} = \frac{\sin \theta}{\theta} = \frac{\theta - \theta^3/3! + \theta^5/5! - \dots}{\theta} = 1 - \frac{\theta^2}{3!} + \frac{\theta^4}{5!} - \dots$$

Equating the previous equations for ℓ/s gives

$$\frac{1}{1 + \epsilon_{TU}} = 1 - \frac{\theta^2}{3!} + \frac{\theta^4}{5!} - \dots$$

This last equation is solved for θ . Then R is given by

$$R = \frac{s}{2\theta} = \frac{1.012 \ell}{2\theta} \quad (4-58)$$

With this value for R determine the allowable lateral pressure loading p_w on the web from

$$p_w = \frac{t F_{TU}}{R} \quad (4-59)$$

The stiffener of length h is assumed to have simply supported ends. The allowable lateral pressure loading p_s on the stiffener is given by

$$p_s = \frac{8I F_{TU}}{c \ell h^2} \quad (4-60)$$

where I is the stiffener moment of inertia, c is the extreme fiber distance to the stiffener outstanding flange from the stiffener centroid, and ℓ , h and F_{TU} are defined above.

Corrugated Web

The analysis procedure used to size the corrugated (120° circular arc) spar webs is presented below. The general and local instability modes of failure under shear loading are considered. The webs within fuel tanks are also sized to preclude failure under lateral (pressure) loading. This lateral loading results from refueling valve malfunction which produces a 9.0 psig ultimate pressure on tank boundary spars and ribs. Lateral loading on intermediate (i.e., nonboundary) fuel tank spars results from the fuel containment requirement for a 6.0 g ultimate aft-acting acceleration (crash condition).

The instability analysis is predicated on the requirement that the chord-to-web attachment maintain the shape of the web corrugation and provide restraint against corrugation warping relative to the chord-to-web attachment plane. These attachment requirements are satisfied by a continuous weld.

Test data for this construction are quite limited; so, consequently, the instability analysis presented herein is considered applicable only for the configuration shown in figure 4-8. The buckling equations are from reference 4-11. The notation used in the analysis is defined as figure 4-8.

The allowable shear stress for general instability is given by

$$\frac{F_s}{E_r} = \frac{K (D/t)^{3/2}}{(h/t)^2} \quad (4-61)$$

where the reduced modulus $E_r = .83E_s + .17E_t$, and the value of K is taken as 1.47 corresponding to the simply supported edge condition.

For the local instability failure mode, the allowable shear stress is given by

$$\frac{F_s}{t_r} = \frac{K}{(D/t)^3 - 2} \quad (4-62)$$

where the reduced modulus $E_r = (EE_t)^{1/2}$, and K is taken as 1.55.

To size the webs for lateral pressure loads, one full-wave of the corrugated web is analyzed as a uniformly loaded fixed-ended beam spanning between the upper and lower chord-to-panel attachment planes. The extreme fiber stress in bending is given by

$$t = \frac{Mc}{I} = \frac{\left(\frac{wL^2}{12}\right)\left(\frac{R+t}{2}\right)}{.544 t R^3} \quad (4-63)$$

where $w = 2 \sqrt{3} Rp$, and $p = 9.0$ psi for refueling overpressurization, or 6.09 psi for the 6.0 g crash condition and 35-inch spar spacing.

Since no test data exists for lateral loading, the allowable extreme fiber stress was conservatively limited to the critical buckling stress for uniform compressive axial loading.

Cellular Core Sandwich Web

The analysis procedure used to size the cellular core sandwich spar webs is presented in this section. For shear loading, the general and local instability modes are considered. The webs within fuel tanks are also sized to preclude failure under lateral (pressure) loading. This lateral loading results from refueling valve malfunction which produces a 9.0 psig ultimate pressure on tank boundary spars and ribs. Lateral loading on intermediate (i.e., nonboundary) fuel tank spars results from the fuel containment requirements for a 6.0 g ultimate aft-acting acceleration (crash condition).

The allowable inplane shear stress (general instability) for a sandwich with isotropic faces of unequal thickness t_1 and t_2 is given by

$$F_s = \frac{G_r}{2 \left[\frac{1}{6} \left(\frac{h}{K_s} \frac{t_1 + t_2}{\sqrt{t_1 t_2}} \right)^2 + \left(\frac{G_r}{2h K_{cs}} \right) \right]} \quad (4-64)$$

where $G_r = 2GG_t/(G+G_t)$, $K_{cs} = 1/2(G_{xz} + G_{yz})$, and K_s is the shear buckling coefficient for the web aspect ratio. Since the web aspect ratios are large, a value of 12.56 is taken for K_s .

The allowable shear intracell buckling stress (local instability) for a sandwich web with isotropic faces is given by

$$F_s = \frac{1.45 \pi^2 G_t}{1 - \mu} \left(\frac{t}{s} \right)^2 \quad (4-65)$$

where s is the diameter of a circle inscribed in a core cell, and t is the thinner of the two face sheets. This allowable is independent of core orientation.

The spar must be further checked for bending resulting from the lateral (pressure) loading. Since the degree of fixity provided by the spar-to-panel joint is difficult to estimate, it has been conservatively assumed that the spar is simply-supported for purposes of checking web bending and fully-fixed for checking bending in the spar-to-panel attachment region.

So for the web, determine the maximum web bending moment per inch, M_i .

$$M_i = \frac{p\ell^2}{8} \quad (4-66)$$

where p is the applied pressure and ℓ is the spar depth. From this moment calculate the maximum stress in each face sheet,

$$f_1 = \frac{M_i}{dt_1}, \text{ and } f_2 = \frac{M_i}{dt_2} \quad (4-67)$$

where $d = h - 1/2(t_1 + t_2)$.

The allowable face sheet stress in tension is the ultimate tension allowable F_{TU} . The allowable face sheet stress in compression is the lowest of the following:

1. Compressive yield stress F_{cy} .
2. Allowable compressive intracell buckling stress which for isotropic faces is given by

$$F_c = \frac{\pi^2 E_t}{3(1 - \mu^2)} \left(\frac{t}{s} \right)^2 \quad (4-68)$$

where s is as defined above and t is the thickness of the face sheet in compression.

Allowable compressive face wrinkling stress is given by

$$F_w = K_w \eta (G_L E_{cc})^{1/3} \quad (4-69)$$

where $\eta = [(3E_t + E_s)/4]^{1/3}$ and $K_w \approx 0.6$. This formula considers failure from core crushing and face separation.

For single edge attachment of the web to the spar chord, the web-attachment leg of the chord is checked for a bending moment per inch, M_i ,

$$M_i = \frac{p\ell^2}{12} \quad (4-70)$$

where p is the pressure and ℓ the spar depth. Since this is an ultimate condition, the bending modulus of rupture, F_b , for the rectangular cross section is used as the allowable. Thus the allowable moment per inch, M_r , is given by

$$M_r = F_b \frac{t^2}{6} \quad (4-71)$$

where t is the thickness of the web-attachment leg. The margin of safety is then given by

$$M.S. = \frac{M_r}{M_i} - 1 \quad (4-72)$$

FUSELAGE PANEL CONCEPTS

The critical design conditions for the fuselage panel concepts were taken from the SCAT-15F study (ref. 4-1) and are shown in figure 2-2. The loads shown are ultimate design loads except for the "1.0 g Cruise-Panel Stability Critical" condition which is indicated on body side and lower centerline control point locations. This design condition was established on the National SST Program to satisfy fatigue and aerodynamic smoothness considerations. Essentially, this design condition requires that "no buckling shall occur at 1.1 times the flight loads existing during steady-state cruise. Actual temperatures and pressures shall be considered."

For the ultimate design conditions, the pressure was conservatively neglected when acting as a relieving load.

Panels bounding the pressurized compartment were designed with an ultimate factor of 2.67 on the maximum operating pressure of 10.78 psi (6400 foot cabin pressure at

65,000 feet). This ultimate load is equivalent to a 2.4 factor on an assumed relief valve pressure of 11.97 psi. This pressure design condition is applicable only at room temperature and is not combined with any other loads.

Panel design for flight loads combined with pressure (if not relieving) used an ultimate factor of 1.67 on the operating pressure differential. For altitudes above 32,300 feet this is equivalent to a 1.5 factor on the relief valve pressure.

Fatigue analysis using rational fatigue load spectra encompassed work beyond the scope of this study. However, experience gained from the National SST Program indicates that 50,000 hours of life can be achieved with a similar body shell if

1. the limit steady-state stresses are restricted to 42 ksi (annealed Ti-6Al-4V), and
2. structural details are designed to a fatigue quality consistent with that used on the Program.

The panel control points selected for evaluation of the fuselage panel concepts were established to be representative of the broad spectrum of design conditions and load intensities encountered on the fuselage. The critical design conditions for the selected control points are all symmetric conditions. Consequently, shear loads do not exist at the upper and lower fuselage centerline control points and bending (i.e., axial) loads do not exist at the fuselage side-panel control points.

Skin-Stringer

The skin-stringer concept was selected as the baseline concept for comparative evaluation since it is the most prevalent concept in current fuselage design. This concept was selected for the National SST and was also the chosen fuselage panel for the SCAT-15F study. The initial panel sizing for each of the fuselage control points was taken from the SCAT-15F study. Each was then updated to insure an optimum baseline concept which had been designed and analyzed in accordance with the ground rules established for this study.

Stringer spacings selected were the result of local panel stringer spacing optimization as well as the practical requirements associated with the required stringer continuity along the length of the fuselage.

The minimum skin gage was established as 0.03 inches on the basis of pressurized cabin criteria and manufacturing handling considerations.

Sizing for ultimate tension loads is based on

$$f_t = \left(\frac{N_t b}{A_g} + \frac{p^R}{2A_g} \right) \frac{1}{K_{NA}} \quad (4-73)$$

where

$$A_g = A_s + bt$$

$$A_s = \text{stringer area}$$

$$b = \text{stringer spacing}$$

$$t = \text{nominal skin gage}$$

$$R = \text{fuselage radius}$$

$$K_{NA} = \text{net area efficiency factor}$$

$$d = \text{frame spacing}$$

$$N_t = \text{ultimate tensile load intensity}$$

Sizing for ultimate compression loads is based on the procedure outlined below:

1. The crippling stress, F_{cc_i} , is established for each element of the stringer from curves similar to that shown in figure 4-9.
2. The stringer crippling stress is calculated as the area-weighted average of the n stringer element crippling stresses, i.e.,

$$F_{cc_{avg}} = \frac{\sum_{i=1}^n (F_{cc_i} A_i)}{\sum_{i=1}^n A_i} \quad (4-74)$$

3. The effective width of buckled skin is calculated as

$$2w = 1.7t \sqrt{\frac{F_c}{F_c}} \quad (4-75)$$

where F_c is the estimated allowable compressive stress.

4. The area and moment of inertia of the effective skin and stringer section is calculated.

5. Assuming simply supported ends at adjacent frames, the effective section slenderness ratio is calculated. Using the normalized Johnson-Euler curves of figure 4-10, the ratio of the allowable column stress to the allowable crippling stress (f_c/F_{cc}) is determined.
6. Using the ratio F_c/F_{cc} and the crippling stress $F_{cc_{avg}}$ from 2. above, the allowable column stress F_c is obtained.
7. The F_c is then compared to the minimum crippling stress F_{cc_i} for any stringer element and the lower is taken as the allowable column stress. Generally, if the minimum crippling stress F_{cc_i} for any stringer element governed, the stringer section was modified to a more efficient geometry.
8. Having established the allowable column stress, the effective skin width was re-calculated and the subsequent procedure repeated until the procedure converges.
9. The allowable column stress was never permitted to exceed the compressive yield strength.

Sizing to preclude compressive buckling for the "1.0 g Cruise-Panel Stability Critical" condition was performed as shown below:

1. With the appropriate geometric parameters and a value of 0.3 for Poisson's ratio, the buckling coefficient K was determined from figure 4-11.
2. For the 1.0 g cruise condition the internal pressure is either 10.78 psi or zero depending upon the panel location. When the pressure is zero, simply supported edges are assumed. For 10.78 psi pressure, fixed edges are assumed and the increase in axial-compressive buckling stress is determined from figure 4-12.

The fuselage side panel sizing procedure is outlined below. As noted earlier, (1) the pressure was conservatively neglected when acting as a relieving load and (2) no bending stresses exist in the side panels. Thus, the ultimate design load is shear acting singly. The analysis that follows is applicable for intermediate shear webs (i.e., a shear web which buckles between 2% and 100% of the design ultimate load). That portion of the shear in excess of the buckling load is carried by diagonal tension in the web and compressive loads in the frames and stringers. The panel buckling stress is given as the equivalent flat panel value plus the increased allowable resulting from curvature.

$$F_{s_{crc}} = F_{s_{crf}} + F_{s_{crs}} \quad (4-76)$$

where

$$F_{s_{crf}} = K_s E_c (t/b)^2$$

in which K_s is taken as 5.25, and

$$F_{s_{crs}} = \frac{\pi}{4} \left(\frac{R_t}{d^2} \right)^{1/4} E_c \left(\frac{t}{R} \right) \quad (4-77)$$

The compressive stress in the stringer resulting from the diagonal tension is given by

$$f'_c = \frac{\gamma \left(f_s - \lambda F_{s_{crs}} \right) b t}{A_{st}} \quad (4-78)$$

where $\lambda = 1$ when direct compression is zero

$$\gamma = 1 + \frac{d}{R} \left(\frac{I_{st}}{J_{st}} \frac{t}{b} \right)^{1/4} \quad (4-79)$$

$$f_s = \frac{N_s}{t} \quad (4-80)$$

and; A_{st} , I_{st} and J_{st} are section properties of the stringer only. The radial load components resulting from the diagonal tension in a curved panel are carried by stringer bending but are considered in this analysis as an equivalent compressive stress given by

$$f''_c = \frac{\gamma F_{s_{crf}} d t}{A_{st}} \left(\frac{0.053 b}{d} \right)^{1/3} \quad (4-81)$$

To establish the allowable shear stress as limited by the stringer the following procedure is used;

1. Estimate an allowable shear stress f_s
2. Calculate f'_c and f''_c per above equations

3. Calculate the allowable column stress of the stringer assuming that no skin is effective in conjunction with the stringer and the stringer has an end fixity coefficient of two at the frames (i.e., $L'/\rho = d/\rho\sqrt{2}$).
4. The stringer design is satisfactory when the sum of f_c' and f_c'' is equal to the stringer allowable column stress.

Next, to determine the allowable shear stress as limited by the web;

1. Estimate an allowable shear stress f_s
2. Calculate the parameters $f_s/F_{s_{crc}}$ and $300 \text{ td}/Rb$
3. With these parameters enter figure 4-13 and obtain the diagonal tension factor K.
4. With K and the minimum ultimate tensile strength F_{TU} for the material (at temperature), enter figure 4-14 and determine the allowable gross area flat-web shear stress $F_{s_{all}}$

With the parameters A_{st}/bt and A_{frame}/dt enter figure 4-15 to get the curved web correction factor Δ .

6. The ultimate allowable shear stress for the curved web is then given by

$$F_s = F_{s_{all}} (0.65 + \Delta) \quad (4-82)$$

with the restriction that the ultimate allowable shear stress shall not exceed 0.38 times the material ultimate allowable tensile stress.

7. The above procedure is iterated until the estimated allowable shear stress is equal to that calculated.

Sizing to preclude shear panel buckling for the "1.0 g Cruise-Panel Stability Critical" condition is outlined below. For this design condition the internal pressure is either zero or 10.78 psi depending upon the panel location. When the pressure is zero, simply supported edges are assumed and the panel buckling stress is given by

$$F_{s_{crc}} = K_s E_c \left(\frac{t}{b} \right)^2 + \frac{\pi}{4} \left(\frac{R_t}{d^2} \right)^{1/4} E_c \left(\frac{t}{R} \right) \quad (4-83)$$

where $K_s = 5.25$. When the internal pressure is 10.78 psi,

1. Clamped edges are assumed from which $K_s = 8.85$

2. An additional capability resulting from the stabilizing pressure is added. With $p/E_c(R/t)^2$ and d/b enter figure 4-16 and obtain ΔC_s . The panel buckling stress is then given by

$$F_{s_{crc}} = 8.85 E_c \left(\frac{t}{b} \right)^2 + \frac{\pi}{4} \left(\frac{Rt}{d^2} \right)^{1/4} E_c \left(\frac{t}{R} \right) + \Delta C_s E_c \left(\frac{t}{R} \right) \quad (4-84)$$

The panel buckling stress given by either of the above equations should not exceed 45% of the material ultimate allowable tensile stress F_{TU} .

Integrally Machined Skin-Stringer

For this concept, shown in figure 2-19, the Ti 6Al-4V skins are machined with longitudinal tabs (short unflanged stiffeners). Angle stiffeners are subsequently welded to these longitudinal tabs. The upper and lower fuselage centerline panels have the stiffeners welded to alternate tabs, while the fuselage side panels have stiffeners welded to each tab.

Tab and stringer spacings selected were the result of local panel stringer spacing optimization as well as the practical requirements associated with the required stringer continuity along the length of the fuselage.

Sizing the ultimate tension loads is based on

$$f_t = \frac{N_t b}{A_g} + \frac{pR}{2A_g} \left(\frac{1}{K_{NA}} \right) \quad (4-85)$$

where

A_g = gross area (skin, tab and stringer)

b = stringer spacing

t = nominal skin gage

R = fuselage radius

K_{NA} = net area efficiency factor

d = frame spacing

N_t = ultimate tensile load intensity

Sizing for compression loads is discussed in this paragraph. The post-buckled strength of the compression panels was less than the buckling strength. Therefore, the ultimate strength was established as the buckling strength and the "1.0 g Cruise-Panel Stability Critical" design condition was necessarily satisfied. The sizing analysis then proceeded in the following steps:

1. For the trial section established, the unflanged tab was evaluated for adequacy by the tab criterion shown in figure 4-17.
2. Next the crippling strength F_{cc} for each element (including the skin segments between the tab and stringers) of the repeating section is determined from curves similar to those of figure 4-9.
3. The section crippling strength is then calculated as the area-weighted average of the n elements of the repeating section.

$$F_{cc_{avg}} = \frac{\sum_{i=1}^n (F_{cc_i} A_i)}{\sum_{i=1}^n A_i} \quad (4-86)$$

4. Then calculate the moment of inertia of the repeating section (tab, skin segments and stringer) about its centroidal axis.
5. The section radius of gyration is then calculated from the section area

$$\sum_{i=1}^n A_i$$

and moment of inertia.

6. Assuming pinned ends at adjacent frames, the slenderness ratio of the section is determined as the frame spacing divided by the section radius of gyration.
7. With the slenderness ratio and the section crippling stress enter the normalized Johnson-Euler curves of figure 4-10 and determine the allowable column stress of the section.

8. The allowable column stress is then compared to the minimum element crippling stress F_{cci} . The lower value is selected as the allowable compressive stress of the section. In general, the optimum design existed when the allowable column stress equaled the minimum element crippling stress and the latter was for the skin segment.

When sizing the integrally machined skin-stringer panels for shear loads, it was considered necessary to preclude panel buckling prior to ultimate load. While no experimental evidence was available, it was hypothesized that unequal tangential load components from the diagonal tension in adjacent panels would result in a torsional instability failure of the stringer. With this premise, the analysis proceeded as below:

1. Based on the buckling equations $F_s = K E_r (t/d)^2$ with the buckling coefficient K taken as 4.9, the minimum skin gage requirement is formulated as

$$t_{\min} = 0.452 d \sqrt{F_s / E_r} \quad (4-87)$$

where the reduced modulus $E_r = \sqrt{E E_t}$

2. The minimum stiffener moment of inertia is based on the criterion from NASA TM 602

$$t_{kr} = \frac{c \sqrt[4]{D_1 D_2^3}}{a^2} \quad (4-88)$$

where

$$t_{kr} = F_{scr} t$$

$$D_1 = \frac{E_{web} t^3}{12 (1 - \mu^2)}$$

$$D_2 = \frac{E_{str} I_{str}}{b}$$

$$a = 1/2 \text{ frame spacing} = d/2$$

Taking the value of c as 8.125 for simply supported edges, introducing the reduced modulus $E_r = \sqrt{E_{web} E_{Tweb}}$ and in accordance with Boeing design practice incorporating an additional factor of safety of 1.5 on the required stiffener moment of inertia, the minimum stiffener moment of inertia is given by

$$l_s = \frac{E_{web}}{E_{str}} \frac{b}{t} \left[\frac{d^2 t}{11.8 E_r} \right]^{4/3} \quad (4-89)$$

Titanium Honeycomb Sandwich

This section outlines the procedures used to size the aluminum brazed titanium honeycomb sandwich fuselage panels. Except for local areas, the core used was SC 4-20 which has a density of 5.0 pcf. This core has a 1/4-inch square cell formed from a foil having a thickness of 0.0020 inch. The core was oriented within the sandwich panels so the ribbon direction is parallel to the longitudinal axis of the fuselage.

Compressive intracell buckling was checked by

$$F_c = K F_t \left(\frac{t}{b} \right)^2 \quad (4-90)$$

where b is the diameter of an inscribed circle within the cell, $K_c = \pi^2/3(1 - \mu^2) = 3.62$ and t is the thickness of the thinner face sheet.

Shear intracell buckling was checked by

$$F_s = K_s E_t \left(\frac{t}{b} \right)^2 \quad (4-91)$$

where b and t are as defined above and $K_s = 5.35[\pi^2/12(1 - \mu^2)] = 4.83$. Compressive face wrinkling was evaluated using the equation

$$F_w = K_w \eta (G_L E_{cc})^{1/3} \quad (4-92)$$

where

$\eta = (3E_t + E_s)/3$, $K_w = 0.6$, G_L is the modulus of rigidity of the core in a plane parallel to the ribbon direction. E is the compressive modulus of elasticity for the core for loads normal to the plane of the core.

Face wrinkling under shear loading was checked by the equation

$$F_{ws} = 0.6 \eta \left[\left(\frac{G_w + G_L}{2} \right) E_{cc} \right]^{1/3} \quad (4-93)$$

Shear crimping failure of the core under compressive end loading was checked by

$$N_{cr} = 0.75 \frac{d^2}{c} G_L \quad (4-94)$$

where N_{cr} is the critical compressive load intensity, c is the core thickness, d is the distance between face sheet centroids and G_L is the core modulus of rigidity previously defined.

The titanium honeycomb sandwich panels were checked for general instability under compressive end loads as a flat, wide-column with ends assumed pinned at the supporting frames. The applicable formula is

$$F_{c_{cr}} = \frac{\pi^2 E_r}{1 - \mu^2} \left[\frac{d \sqrt{t_1 t_2}}{L (t_1 + t_2)} \right]^2 \quad (4-95)$$

where $E_r = 2EE_t/(E + E_t)$ and other symbols are as previously defined. The panels were checked for general instability under shear loading as a curved shear panel with isotropic faces, i.e.,

$$F_s = \frac{G_r}{2 \left[\frac{1}{6} \left(\frac{b}{d \sqrt{K_s}} \frac{t_1 + t_2}{2 \sqrt{t_1 t_2}} \right)^2 + \left(\frac{G_r}{2 h K_{cs}} \right) \frac{t_1 + t_2}{t_1 + t_2} \right]} \quad (4-96)$$

where $K_{cs} = (G_L + G_W)/2$, $G_r = 2GG_t/(t_1 + t_2)$, and other notation is as previously defined above. K_s , the buckling coefficient for simply supported curved shear panels, was read from figure 4-18 wherein the applicable aspect ratio was taken as that of a cylinder.

The fuselage centerline panels were checked for the combined effects of uniaxial longitudinal tensile end load and the applicable internal pressure.

STRUCTURAL CONCEPTS WEIGHT ANALYSIS

The wing skin panels, wing spars, and body shell were chosen as components for evaluation because they have a major impact on the structural airframe weight. For each of these components, control points have been chosen on the wing and body. At these representative locations loads and environmental conditions were established from the 969-336C documentation. For each control location the baseline structure from the 969-336C was established and each alternate concept developed was designed to the same set of baseline conditions. The establishment of specimen configurations was a compromise, and included as many edge member and joint details as possible within the

time limitations of the study. It should be recognized, however, that a total major component weight can only be accurately obtained by a complete design and analysis of all joints and components involved. The concept specimen weight represents an indicator of the relative effectiveness of various concepts, however, these cannot be extrapolated to find total weight differences between concepts for major airframe components.

WING PANEL WEIGHT ANALYSIS

Figure 2-1 identifies the locations and design conditions for the wing panel control points 249, 269, and 431. At each of these locations a control specimen was designed in detail for each structural concept. Figure 2-4 shows a typical example. The control specimen in each case was 12 inches in span and half the spar spacing wide. The total weight of each specimen was calculated and reduced to an average weight per square foot for comparative purposes.

Figure 4-19 shows the relative weight per square foot for each of the specimens without any thermal insulation provisions. Point 249 represents a lightly loaded portion of the wing designed by refuel overpressure. Stressskin and aluminum brazed honeycomb were the only viable contenders in this minimum gage portion of the structure. The stressskin is slightly lighter than the brazed honeycomb. The upper skin panels at points 269 and 431 are designed by biaxial compression plus shear. In each of these cases, the aluminum brazed honeycomb is the minimum weight configuration, followed by the integrally machined and welded concept. The lower surface panel at point 269 is critical for the tension plus shear condition. The stressskin design is the lightest, with the integrally machined and welded panel being second. The lower surface panel at point 431 is likewise critical for tension plus shear. The integrally machined and welded concept is lightest, followed by the aluminum brazed titanium honeycomb sandwich.

WING SPAR WEIGHT ANALYSIS

Spar configurations, as shown in figures 2-5, 2-6, 2-14 through 2-17, have been designed and analyzed for points 249, 269 and 431. At each location a total specimen weight has been calculated and reduced to a weight per linear-foot of spar length for comparative purposes. Figure 4-20 shows the spar weight summary. At each of the locations evaluated, the welded sine wave configuration represents the lightest structural arrangement, followed by the sheet stiffener concepts, with the honeycomb arrangements the least efficient. The welded configuration is the lightest of the sheet stiffener concepts followed closely by the machined extrusion and riveted sheet stiffener configurations. Previous SST studies have established that hole and bracket reinforcements in the welded sine wave construction, greatly reduce the weight efficiency and increase the complexity. This indicates that the quantity of systems routing penetrations in various portions of the wing is a consideration in the final choice of spar concept.

BODY SKIN PANEL WEIGHT ANALYSIS

Figure 4-21 shows the four sections along the body at which ten control points have been established. Three body structural concepts including stiffened sheet, integrally stiffened skin, and honeycomb sandwich have been designed in detail for the baseline structural loads and environmental conditions at the ten control point locations. The aluminum brazed titanium honeycomb shown in figure 2-18 for body application is a typical example. The total weight of each panel specimen was calculated and reduced to an average weight per square foot for comparative purposes.

A summary of the resulting weights for the three body structural concepts is given in figure 4-21. The brazed titanium honeycomb concept is slightly lighter than the riveted sheet stiffener at the forward body section where the loads are relatively low and at isolated locations critical for compression. In nearly all of the more heavily loaded body sections the sheet stiffener is very competitive with the brazed honeycomb unit weights. The integrally machined sheet stiffener panels were consistently heavier than other structural arrangements.

MANUFACTURING PRODUCIBILITY AND MATERIAL COST ANALYSIS

Each of the structural concepts has been evaluated for manufacturing producibility and compared to a baseline structural concept for complexity. The baseline arrangements have arbitrarily been rated at 100 and increasing complexity is given an increasing number. For example, a configuration which was evaluated to be twice as difficult to fabricate as the baseline would be rated 200. The following evaluation ground rules were used for the study:

- Assume a production program of 200 airplanes
- Engineering go-ahead for the program is 1975
- Capital facilities and plant equipment, raw materials and manufacturing and inspection processes development requirements are excluded from the manufacturing producibility ratings
- The wing panels have gentle compound contours
- Ratings give consideration to both nonrecurring and recurring requirements for in-house Boeing effort

WING PANEL PRODUCIBILITY RATINGS

The wing panel producibility ratings for each structural concept is given in table 4-1 with the corrugated core sandwich used as a baseline and rated as 100. The concept complexities range from the least complex of 54 for sheet stiffener to 189 for brazed steel. A pattern to the producibility ratings can be observed. Those concepts made by conventional practices including machining, forming, and riveting are rated most favorable. Those concepts utilizing welding for assembly tend to be of intermediate

complexity. Configurations requiring brazing tend to be the most complex because of the close fit up requirements as well as complexity of the brazing process.

WING SPAR PRODUCIBILITY RATINGS

The wing spar producibility ratings are shown in table 4-2. The riveted sheet stiffener construction is used as a baseline for reference and given a producibility rating of 100. The alternate types of construction tend to follow the same complexity trends noted for the wing panels. The simplicity of riveted sheet stiffener gives it the best producibility rating, followed by the welded concepts, with the brazed concepts being rated the most complex. One notable exception to the trend is the sheet stiffener machined extrusion configuration. This configuration is made by conventional processes, machining and riveting, but is rated more complex than some welded configurations because of the extensive machining required. The minimum extrusion gage of .44 inch requires that every surface of both stiffeners and web be extensively machined.

BODY PANEL PRODUCIBILITY RATINGS

The pattern of body structural concept producibility ratings is similar to the wing panel and spar evaluation. The producibility spread shown in table 4-3 ranges from 100 for the baseline riveted sheet stiffener concept to 380 for corrugated core sandwich. The baseline sheet stiffener arrangement utilizes machined skins and formed sheet metal stiffeners. Another sheet stiffener configuration, utilizing machined skins and machined extrusions for stiffeners, is rated at 158. The producibility rating of 380 for corrugated core sandwich is due to the complex compound contour forming required for the core fabrication and to the welding fit up requirements for these body skin panels.

MATERIAL COST ANALYSIS

This analysis was initiated to find the impact of the various fabrication processes utilized in the structural concept designs on material costs. Figure 4-22 shows the range of minimum (cross-hatched) and maximum material cost lb of finished part for each construction type. Except for aluminum brazed titanium body panels and the stressskin wing panels, the material cost lb of finished part falls within a relatively narrow band. The wide range of material cost on the body skin panels is due to the extensive gage change required on long body skins. The initial material gage must be adequate for the heaviest splice joint and in many cases is tapered to a minimum gage section. This extensive taper on relatively thin gages means a high percent of total metal removal, which increases the material cost lb of finished part. The high cost per lb of finished part for stressskin is caused by the extensive fabrication processes involved in the spot welding assembly of the honeycomb by the subcontractor before the material is purchased.

In the selection of construction types for the 969-512B it would not appear that the cost lb of finished part would be a major influencing factor in concept selection with the exception of brazed body skin panels and the application utilizing stressskin.

STRUCTURAL CONCEPTS EVALUATION AND SELECTION

The wing and body structural concepts have been evaluated on the basis of critical characteristics, and selections have been made. The structural concepts selected from the evaluation were reviewed for application to the 969-512B. For the most part the selection was based on the formal evaluation; however, in a few locations overriding practical requirements necessitated a structural configuration other than that selected through the evaluation process.

EVALUATION CRITERIA

The initial screening of structural concepts considered the following ground rules:

- No structural concept was evaluated which had a major defect which prevented it from being a competitive candidate in all major requirements.
- Structural recommendations have been revised from the theoretical evaluation recommendations when overriding practical requirements dictate a particular construction type.
- Candidate structural types evaluated have been selected based on previous national SST evaluation experience.

The structural concepts evaluation utilized a rating system which considered the characteristics of weight, manufacturing complexity, stiffness, maintenance, fatigue (DFR), fail safety, thermal conductance, and material cost. As seen in tables 4-4 through 4-7, a rating factor was selected for each characteristic from a range of 0 to 100 (very poor to very good) based on the concept's merit relative to a baseline which was arbitrarily given 50. Construction types were screened through several levels of evaluation. Each level of evaluation considered a different characteristic starting with the most important (weight) and continuing in a descending order of importance to the lowest (material cost). Starting with two or three candidates from the first screening level, the best were evaluated through enough levels to establish a superior candidate. The symbols are for those concepts selected to carry through to another level of evaluation until a superior candidate is established which is noted by $\square$.

CONCEPTS EVALUATED

Structural concepts have been evaluated at three locations on the wing and four body stations (ten locations). Those concepts which were evaluated are shown in table 4-8. It should be pointed out that Stresskin was eliminated from consideration because of unacceptable risks due to manufacturing and durability. The problem of designing and manufacturing an acceptable edge attachment on the Stresskin panels was never satisfactorily resolved during the National SST Program, in spite of considerable engineering effort. Also, the use of spot welding of the core and face sheets raised the question of durability that has never been satisfactorily answered. Boeing's experience with spot welds in titanium has been very unsatisfactory.

SCREENING ANALYSIS

The detailed evaluation analysis for selection of structural concepts is shown in the following tabular charts:

Table No.	Component	Location
4-4	Wing Upper Surface Panel	269 UPR
	Wing Lower Surface Panel	269 LVR
4-5	Wing Upper Surface Panel	431 UPR
	Wing Lower Surface Panel	431 LWR
4-6	Wing Spar	PT 269, 431 and 249
4-7	Body Panel Structure	Control Points (1) through (10)

STRUCTURAL CONCEPT SELECTION FOR THE 969-512B (1975 Technology)

The results of the concept evaluation and final structural selection are shown in table 4-9 and figure 4-23. It should be noted that the structural recommendation for application to the 969-512B is not always the same as the evaluation selection. When this occurs, the reason for deviation is given in table 4-9.

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- 4-4 Gerard, G., "Minimum Weight Analysis of Orthotropic Plates Under Compressive Loading" *Journal of the Aeronautical Sciences*, January 1960, pp. 21-26 and 64.

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- 4-10 Anderson, M. S., *Local Instability of the Elements of a Truss-Core Sandwich Plate*, NASA TN-4292, Langley Aeronautical Laboratory, Langley Field, Virginia, July 1958.
- 4-11 Corrugated Shear Web Allowables, p. 3-1-50-1, North American Aviation, Inc., Structures Manual, April 12, 1962.

Table 4-1.—Wing Panel Producibility Ratings

Wing skins	Rating
Riveted Ti	54
Machined and welded (long stiffeners)	84
Machined and welded (X-sect stiffener)	94
Corrugated core	100 ⁽¹⁾
Integrally stiffened	125
Stresskin	130 ⁽²⁾
Al brazed Ti H/C	179
Brazed steel H/C	189

(1) Baseline

(2) Boeing in-house effort only

Table 4-2.—Wing Spar Producibility Ratings

Wing spars	Rating
Riveted web and stiffener	100 ⁽¹⁾
Sine wave	157
Machined and welded web stiffener	167
Extruded web-stiffener	170
Al brazed H/C (double upr and lwr chord)	276
Al brazed H/C (single upr and lwr chord)	289
Brazed steel H/C (double upr and lwr chord)	350
Brazed steel H/C (single upr and lwr chord)	374

(1) Baseline

Table 4-3.—Body Panel Producibility Ratings

Body structure	Rating
Riveted skin:str (sheet)	100 ⁽¹⁾
Riveted skin:str (machined)	158
Welded skin:stiffener	172
Brazed Ti H/C (alternate no. 1)(fastened frame)	365
Corrugated core sandwich	380

(1) Baseline

Table 4-4.-Structural Concept Evaluation, Wing Skin Panels, Point 269

Component*	Item	Rating range	Baseline (corr core sandwich)	Al brazed T ₁ H/C	Brazed steel H/C	Sheet stiff T ₁	Integrally mach and weld sheet-stuff	Integrally mach waffle
Upr surface	Weight	0-100 ←————→	50	69	29	50	63	44
	Mfg complexity		50	40			52	
	Stiffness		50	53			50	
	Maintain		50	70			80	
	Fatigue (DFR)		50					
	Fail safety		50					
	Thermal conductance		50	27			1	
	Material cost		50					
Lwr surface	Weight	0-100 ←————→	Baseline (sheet-stuff)	56	10		57	
	Mfg complexity		50	33			45	
	Stiffness		50	60			62	
	Maintain		50	35			50	
	Fatigue (DFR)		50					
	Fail safety		50					
	Thermal conductance		50					
	Material cost		50					



Carry through to another level

A superior candidate established on basis of factors considered

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Table 4-5.—Structural Concept Evaluation, Wing Skin Panels, Point 431

Component	Item	Rating range	Baseline (corr core sandwich)	Al brazed Ti H/C	Brazed steel H/C	Sheet stiff Ti	Integrally mach and weld sheet-stiff	Integrally mach waffle
Upr surface	Weight	0-100 ↑ ↓	50	68	47	30	56	20
	Mfg complexity		50	40			52	
	Stiffness		50	57			55	
	Maintain		50					
	Fatigue (DFR)		50					
	Fail safety		50					
	Thermal conductance		50					
	Material cost	0-100	50					
Component	Item	Rating range	Baseline (corr core sandwich)	Al brazed Ti H/C	Brazed steel H/C	Sheet stiff Ti	Integrally mach and weld sheet-stiff	Integrally mach waffle
Lwr surface	Weight	0-100 ↑ ↓	50	49	17	38	58	47
	Mfg complexity		50	40			52	
	Stiffness		50	55			55	
	Maintain		50	70			80	
	Fatigue (DFR)		50	58			64	
	Fail safety		50					
	Thermal conductance		50					
	Material cost	0-100	50					



Carry through to another level



A superior candidate is established on basis of factors considered

Table 4-6.—Structural Concept Evaluation, Wing Spar

Location	Item	Rating range	Baseline welded sine wave	Riveted sheet stiff	Welded sheet stiff	Extruded sheet stiff	Brazed Ti H/C
Pt 269	Weight	0-100 ↑ ↓	50	25	32.5	26.2	16.5
	Mfg complexity		50		48		
	Maintain		50		60		
	Fatigue (DFR)		50		46		
	Thermal conductance		50				
	Material cost	0-100	50				
Pt 431	Weight	0-100 ↑ ↓	50	33	40.5	37.5	22.5
	Mfg complexity		50		48	47	
	Maintain		50		50	55	
	Fatigue (DFR)		50		46	32	
	Thermal conductance		50				
	Material cost	0-100	50				
Pt 249	Item	Rating range	Baseline riveted sheet stiff	Welded sine wave	Welded sheet stiff	Extruded sheet stiff	Brazed Ti H/C
	Weight	0-100 ↑ ↓	50	58	57	56	23
	Mfg complexity		50	43	42	40	
	Maintain		50	40	40	45	
	Fatigue (DFR)		50	68	64	50	
	Thermal conductance		50	70	60		
	Material cost	0-100	50				



Carry through to another level



A superior candidate is established on basis of factors considered

Table 4-7. - Structural Concept Evaluation, Body

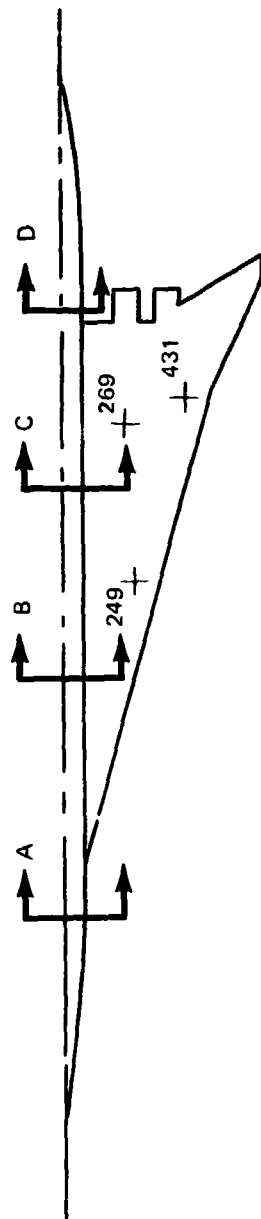
Location	STA 1005			STA 1720			STA 2474			STA 3100		
	(1) Crown	(2) Side	(3) Crown	(4) Side	(5) Keel	(6) Crown	(7) Side	(8) Crown	(9) Side	(10) Keel	(0-100)	
Rating range	(0-100)											
Weight	Sheet stiff (baseline) integrally mach T ₁ H/C	50 43 52	50 38 56	50 15 50	50 43 61	50 38 54	50 22 67	50 50 77	50 21 58	50 34 57		
Mfg complexity	Sheet stiff (baseline) integrally mach T ₁ H/C	50 28	50 28	50 28	50 33	50 33	50 33	50 48 33	50 50 28	50 50 33		
Maintain	Sheet stiff (baseline) integrally mach T ₁ H/C	50 40	50 40	50 40	50 40	50 40	50 40	50 45 40	50 43 28	50 40		
Fatigue (DFR)	Sheet stiff (baseline) integrally mach T ₁ H/C	50	50	50	50	50	50	50	50	50		
Fail safety	Sheet stiff (baseline) integrally mach T ₁ H/C	50	50	50	50	50	50	50	50	50		
Material cost	Sheet stiff (baseline) integrally mach T ₁ H/C	50	50	50	50	50	50	50	50	50		

☐ Carry through to another level

☒ A superior candidate is established on basis of factors considered

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Table 4-8.—Structural Concepts Evaluated



Component	Location	Material	Concept
Wing skin panels	269, 431	Ti	Corrugated core sandwich
	249, 269, 431	Ti	Al brazed Ti H/C
	269, 431	Steel	Brazed steel H/C
	269, 431	Ti	Sheet stiffener
	269, 431	Ti	Integrally machined sheet stiffener
Wing spars	269, 431	Ti	Integrally machined waffles
	249, 269, 431	Ti	Welded sine wave
		Ti	Riveted sheet stiffener
		Ti	Welded sheet stiffener
		Ti	Extruded sheet stiffener
Body skin covers	A, B, C, and D	Ti	Brazed Ti H/C
			Sheet stiffener
			Brazed Ti H/C
			Integrally machined

Table 4-9.—Structural Concept Recommendations 969-512B

Component	Location	Evaluation recommendation	969-512B structural recommendation	Comments
Wing upr surface	Entire in-spar area	Al brazed Ti H/C	Al brazed Ti H/C	Combined body bending and wing loads require high biaxial capability compatible with sheet stiffener construction
Wing lwr surface (pt 249)	See fig. 4-23 ↑	Al brazed Ti H/C	Al brazed Ti H/C	
Wing lwr surface (pt 269)		Ti integrally mach and welded sheet stiffener	Ti integrally mach and welded sheet stiffener	
Wing lwr surface (center section)		-----	Integrally mach waffle	
Wing lwr surface (pt 431)	See fig. 4-23 ↓	Ti integrally mach and welded sheet stiffener	Ti integrally mach and welded sheet stiffener	Equipment installation requires cutout reinforcements, which are impractical for sine wave
Wing spar (pt 249 and 431)		Ti welded sine wave	Ti welded sine wave	
Wing spar (pt 269)		Welded sine wave	Ti sheet stiffener	
Body shell	See fig. 4-23	Ti riveted sheet stiffener	Ti riveted sheet stiffener	

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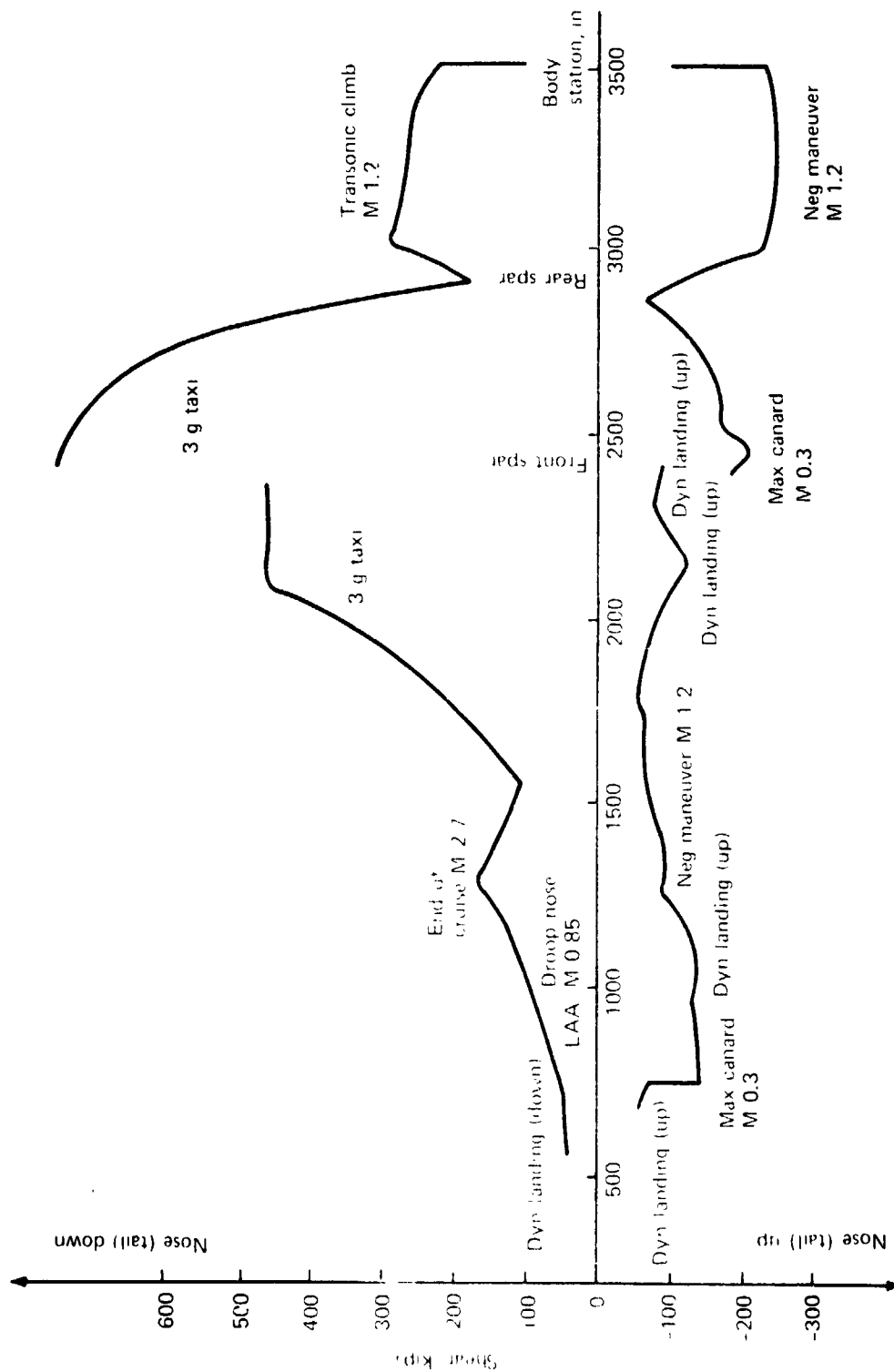


Figure 4.1.—Body Shear, 969-336C

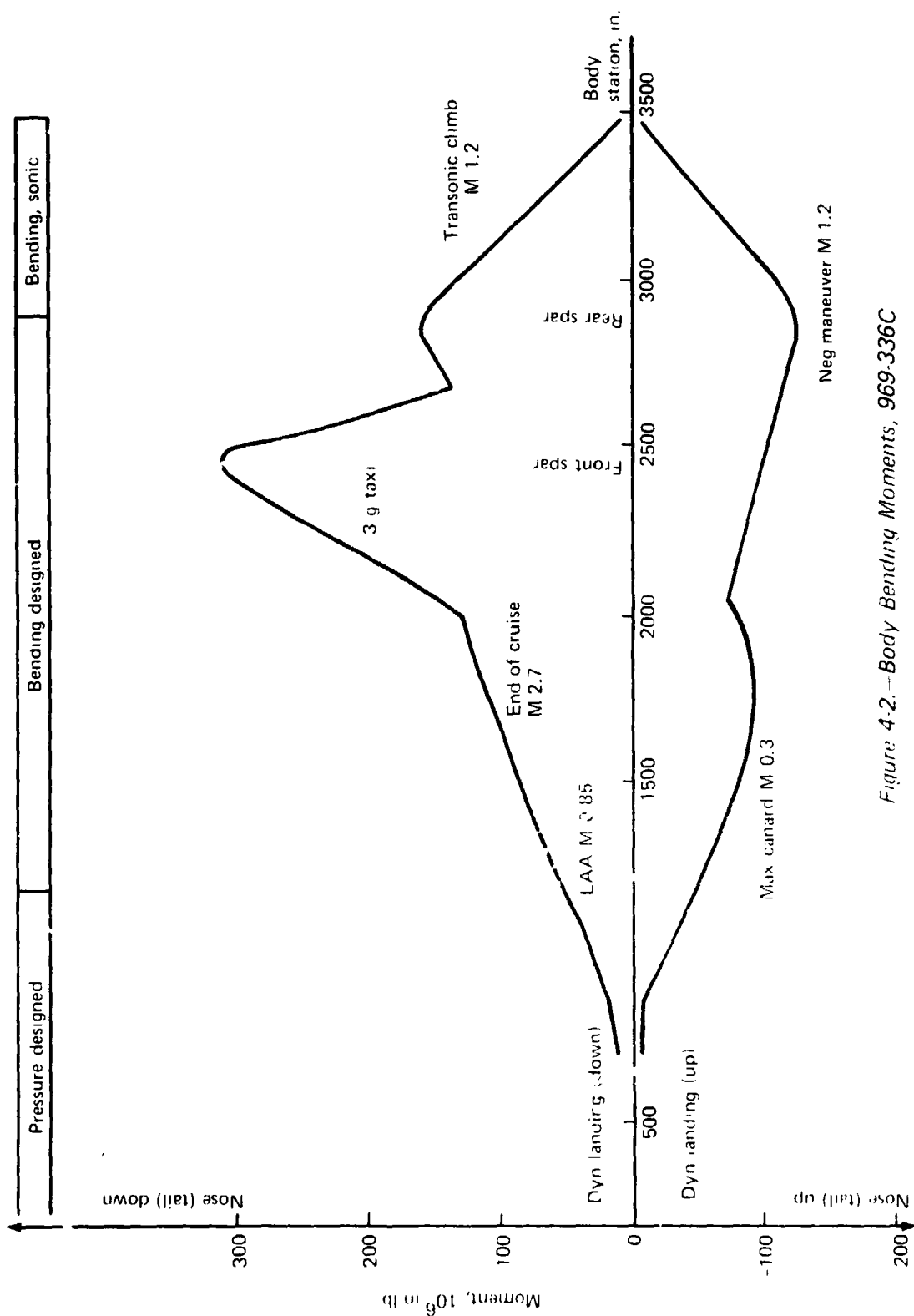
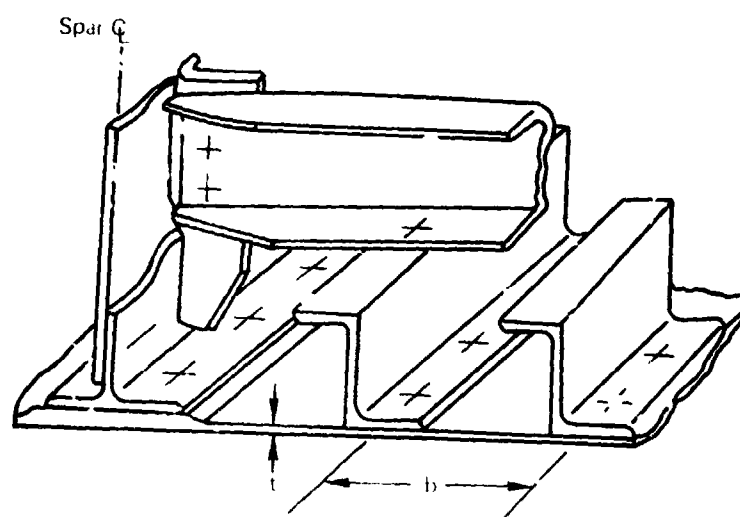
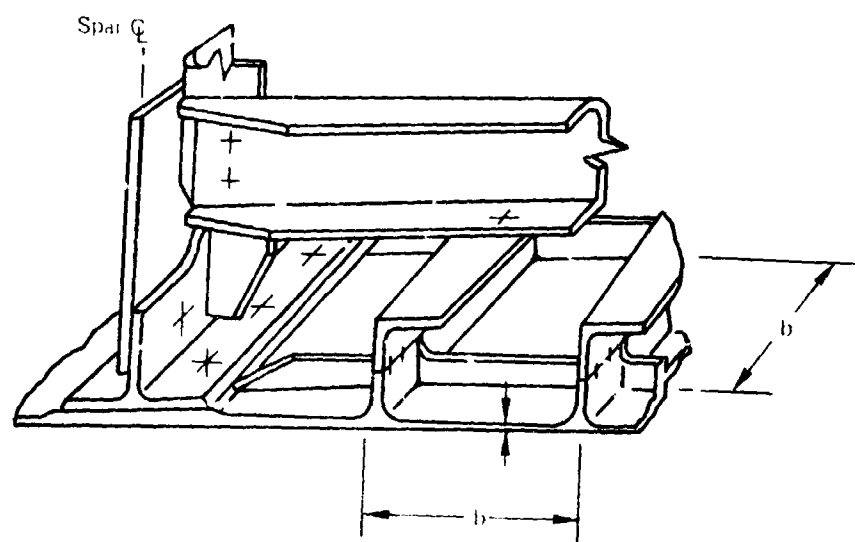


Figure 4-2.—Body Bending Moments, 969-336C

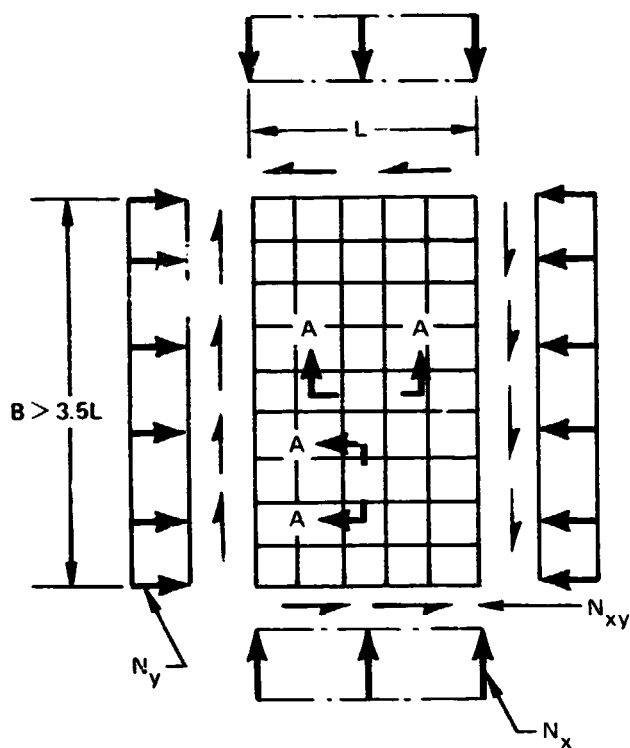


Riveted



Integral

Figure 4.5 Skin-Stringer Concepts



Simply Supported
Compression Panel

$$\begin{aligned} r_t &= t_w/t_s \\ r_b &= b_w/b_s \\ \bar{t}/t_s &= 1 + 2r_b r_t \\ KA &= (\bar{t}/t_s) = 1 + r_b r_t \end{aligned}$$

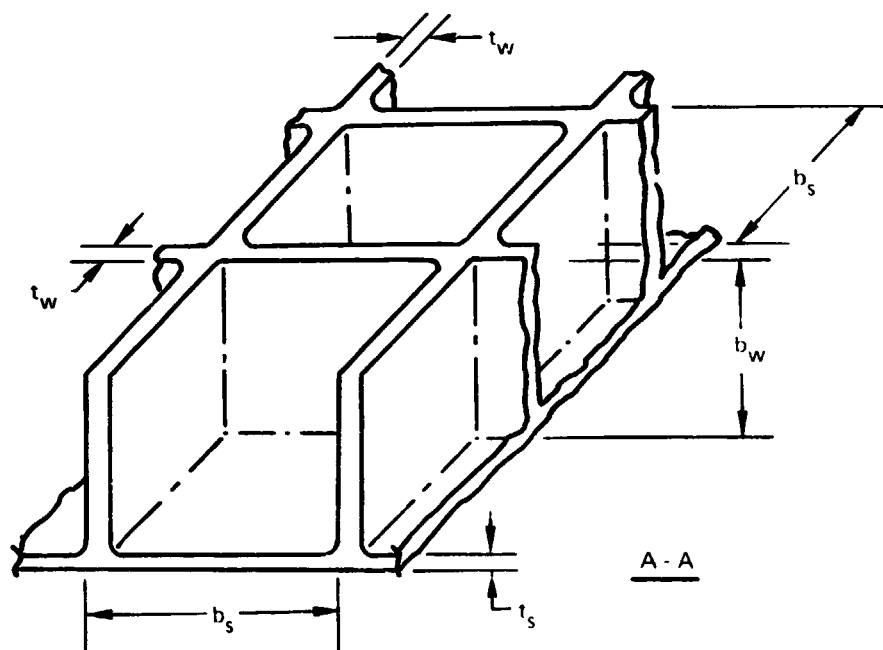


Figure 4-4.—Integrally Machined Waffle

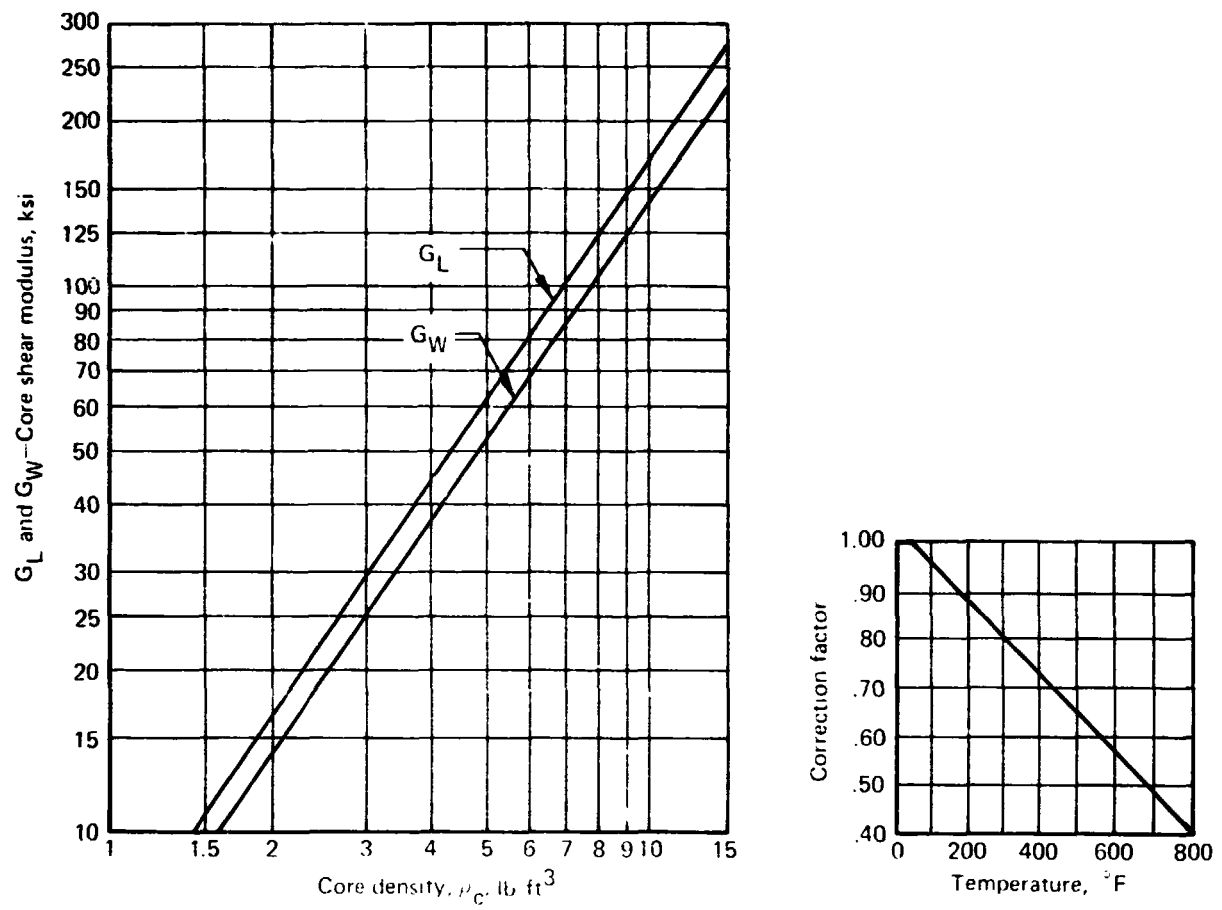


Figure 4-5.—Core Shear Modulus

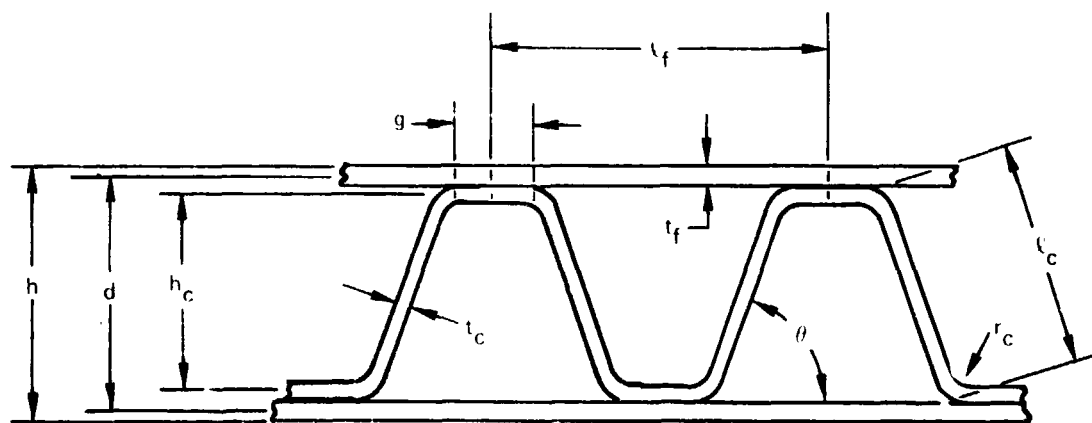


Figure 4-6.—Corrugated Core Sandwich

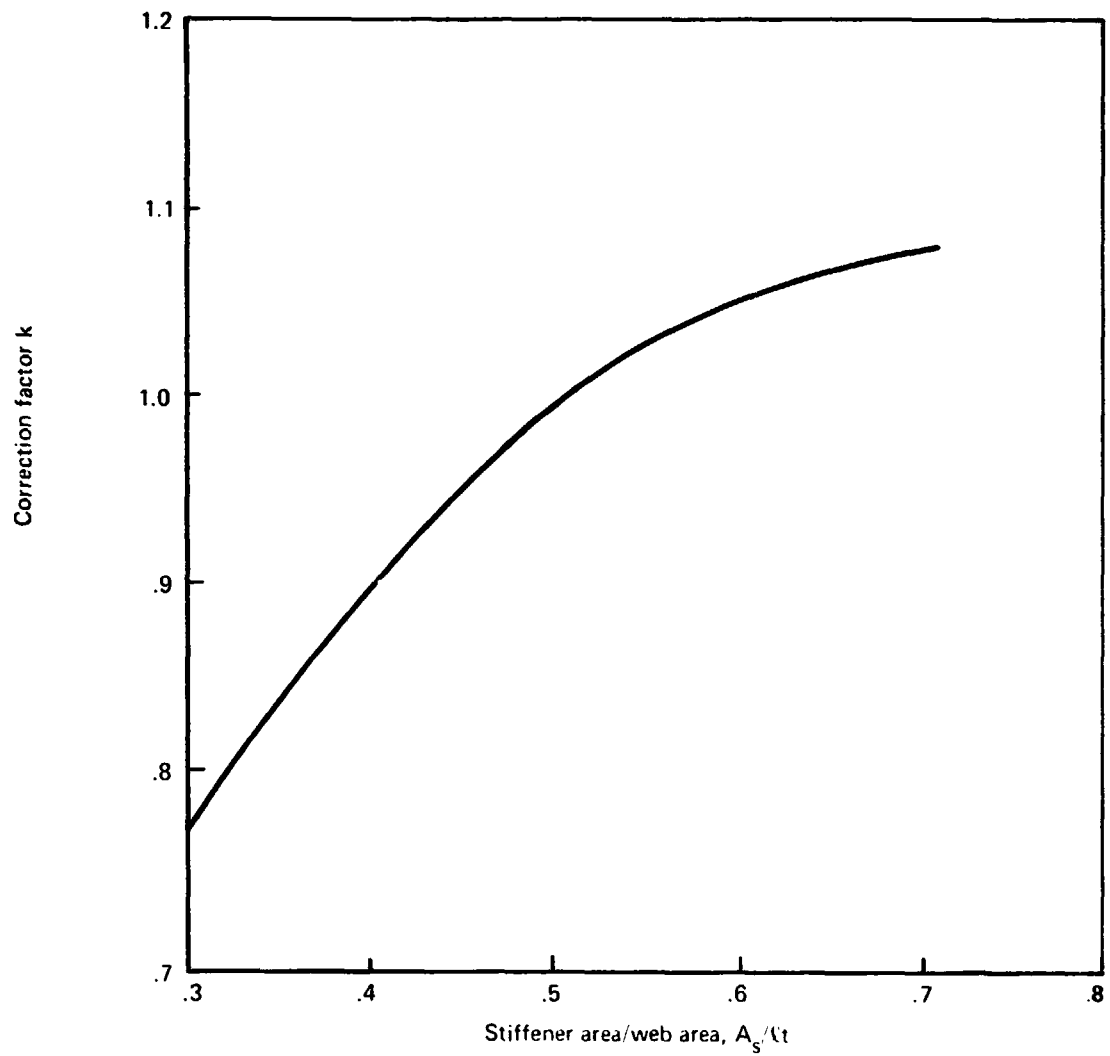


Figure 4-7.—Shear Flow Correction Factor

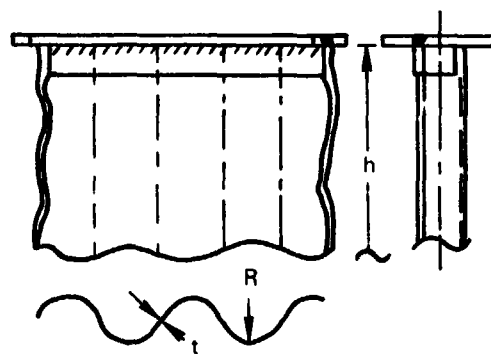
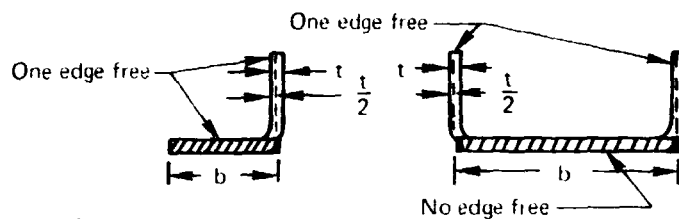
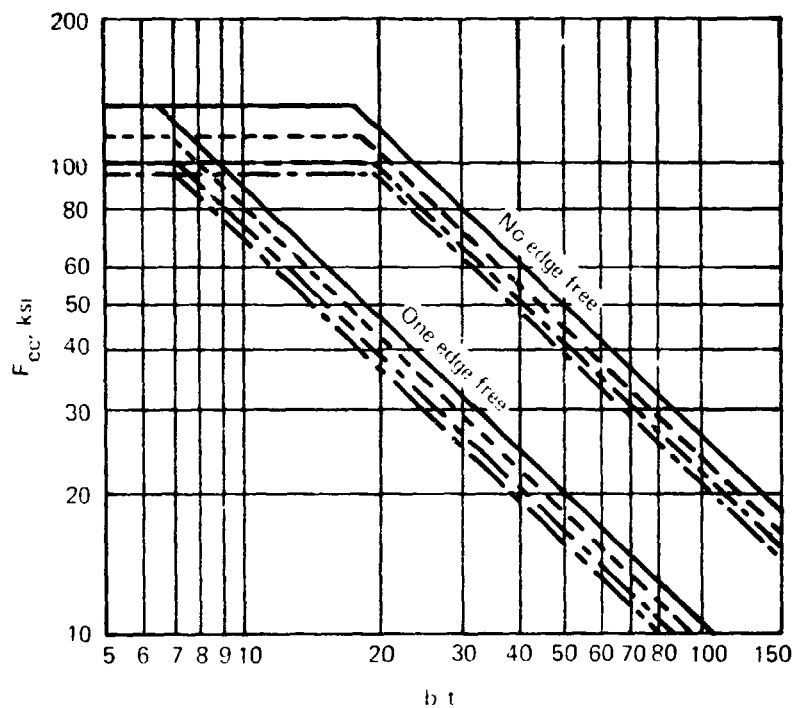


Figure 4-8 —Corrugated Web, Spar Configuration



$$F_{cc} = \frac{\sum b_n t_n F_{ccn}}{\sum b_n t_n}$$

Ti-6Al-4V (MIL-T-9046)
Annealed
Sheet and plate



- 70 $F_{cy} = 132\,000$ psi
- 250 $F_{cy} = 114\,000$ psi
- · - · - 400 $F_{cy} = 101\,000$ psi
- · — · — 500 $F_{cy} = 91\,300$ psi

Figure 4-9 —Compressive Crippling of Formed Sections

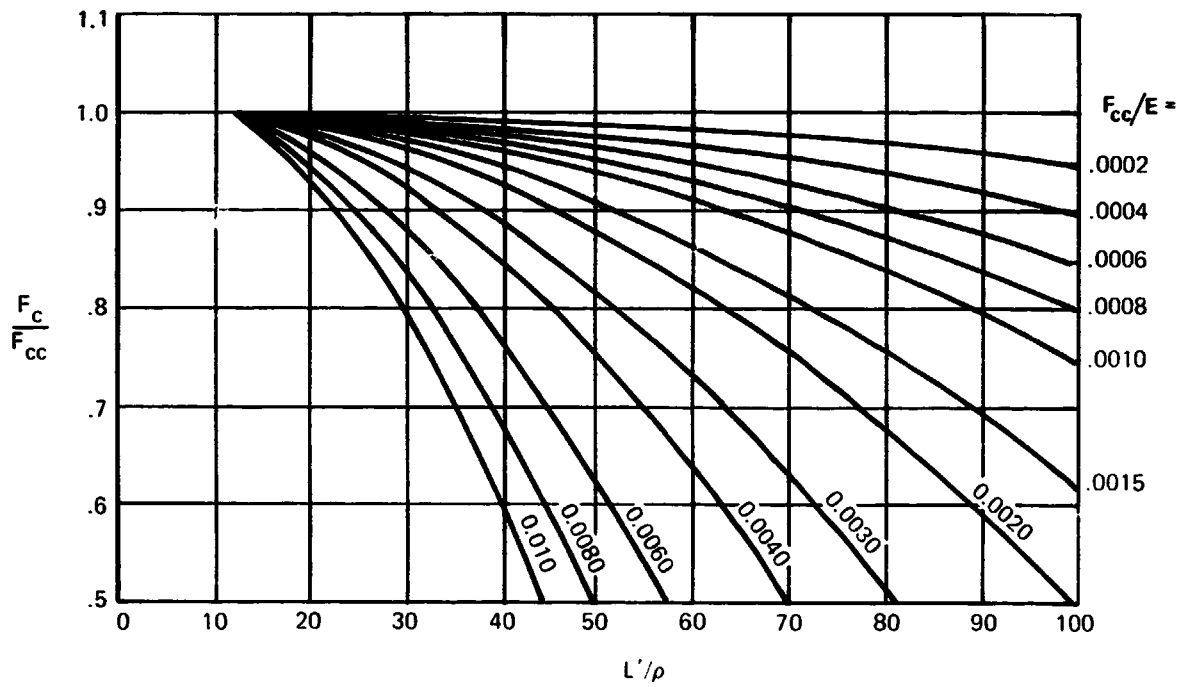
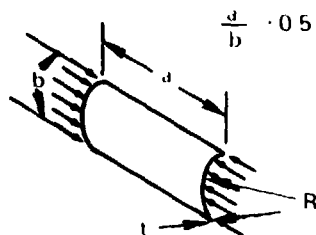


Figure 4-10.—Normalized Johnson-Euler Curves



$$\frac{\sigma_{cr}}{\eta} = K \frac{\pi^2 E}{12(1-\mu^2) \left[\frac{1}{b} \right]^2}$$

$$Z = \frac{b^2}{Rt} \sqrt{1-\mu^2}$$

— Simply supported edges
 - - - Clamped edges

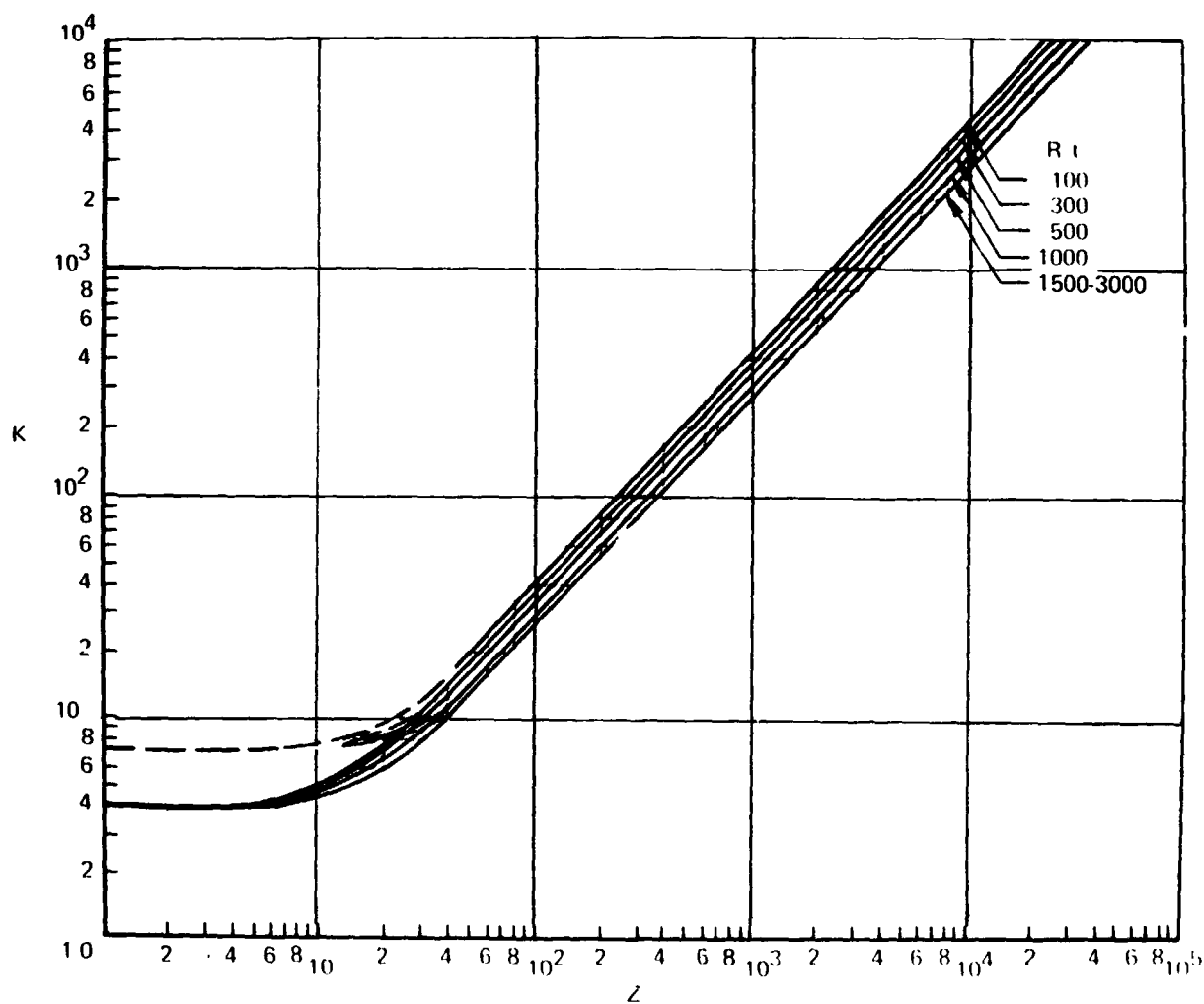


Figure 4-11 Buckling Stress Coefficient, K_C , for Unpressurized Curved Panels Subjected to Axial Compression

$$\Delta\sigma_{cr} = \Delta C_c E_c t/R$$

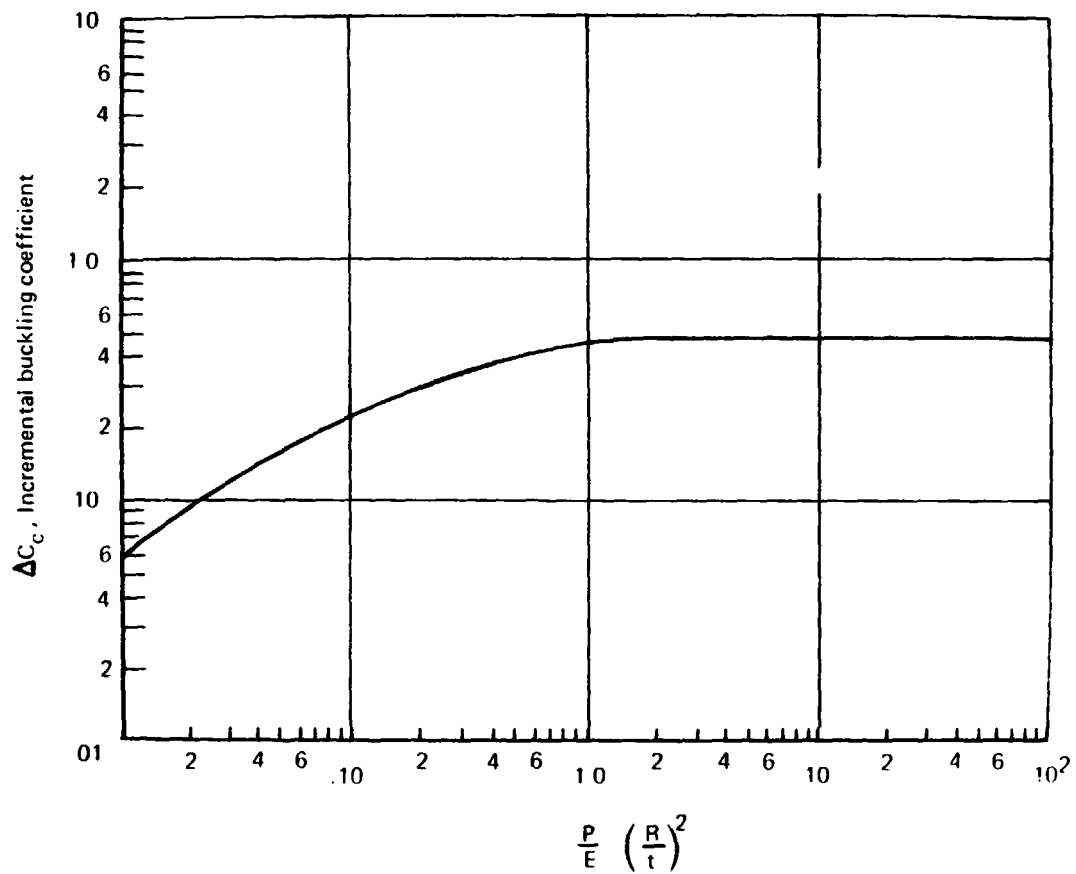


Figure 4-12.—Increase in Axial-Compressive Buckling Stress of Curved Panels Due to Internal Pressure

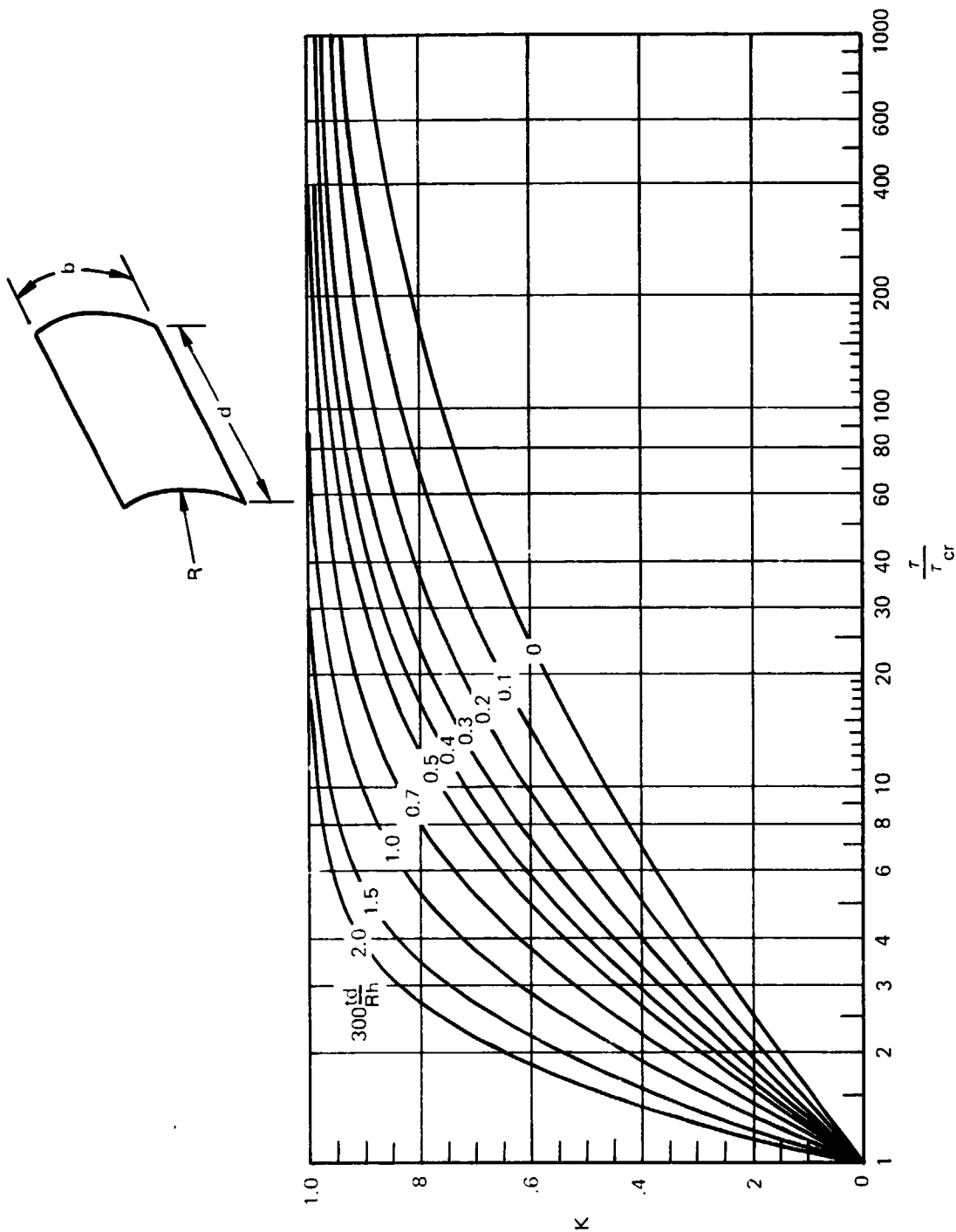


Figure 4-13.—Diagonal—Tension Factor, K , (if $\frac{d}{b} > 2$, use 2)

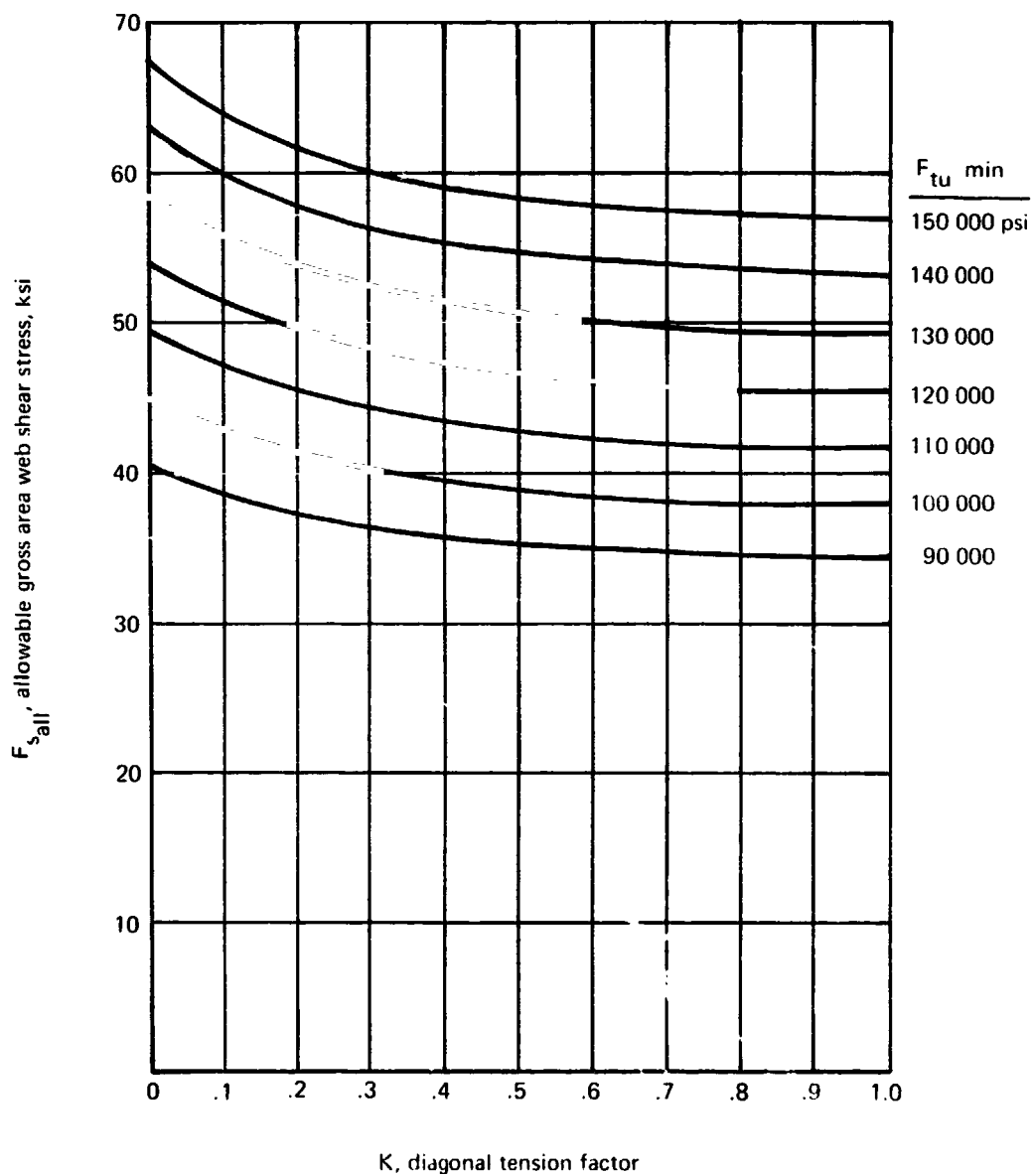


Figure 4-14.—Allowable Gross Area Web Shear Stress (Without Curvature Correction)

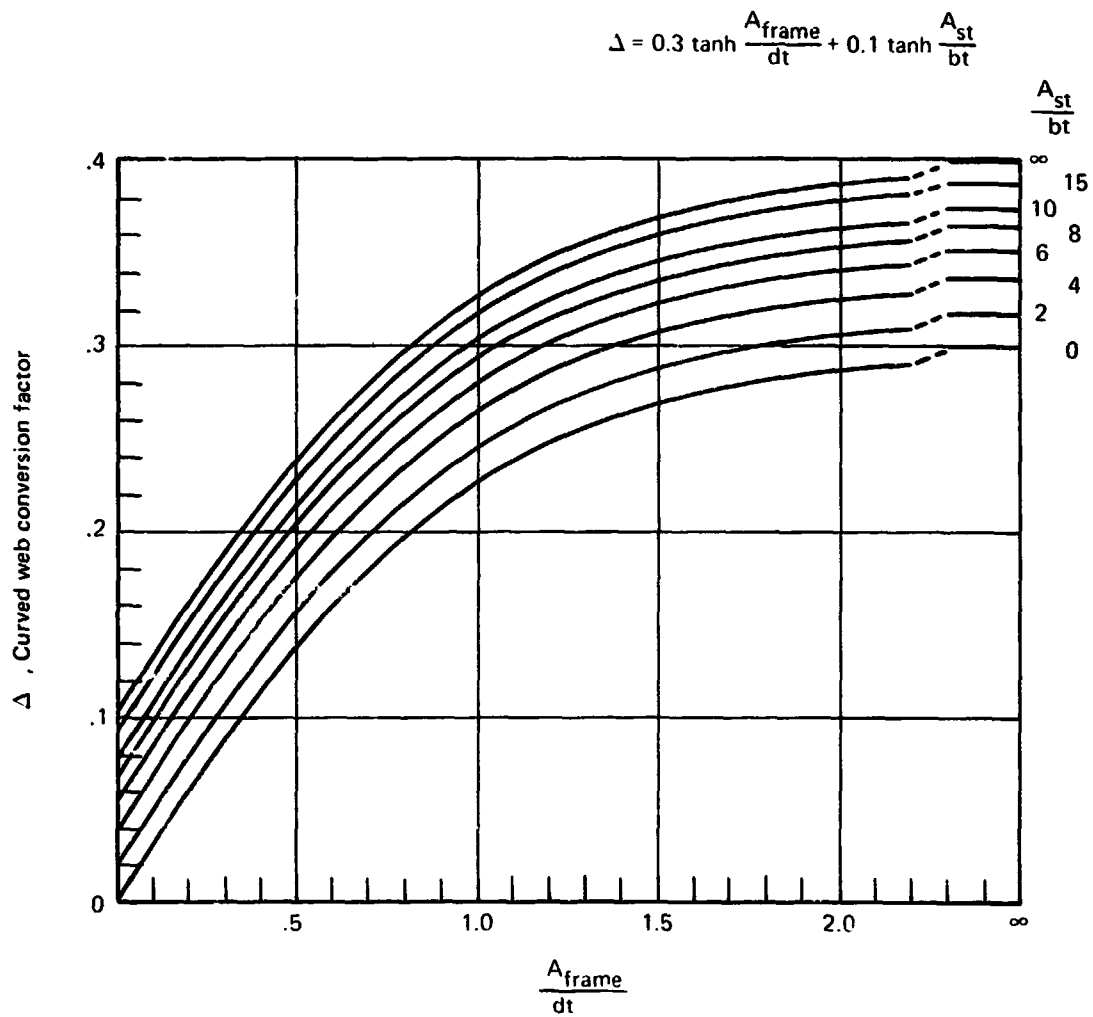


Figure 4-15.—Correction for Allowable Ultimate Shear Stress in Curved Webs

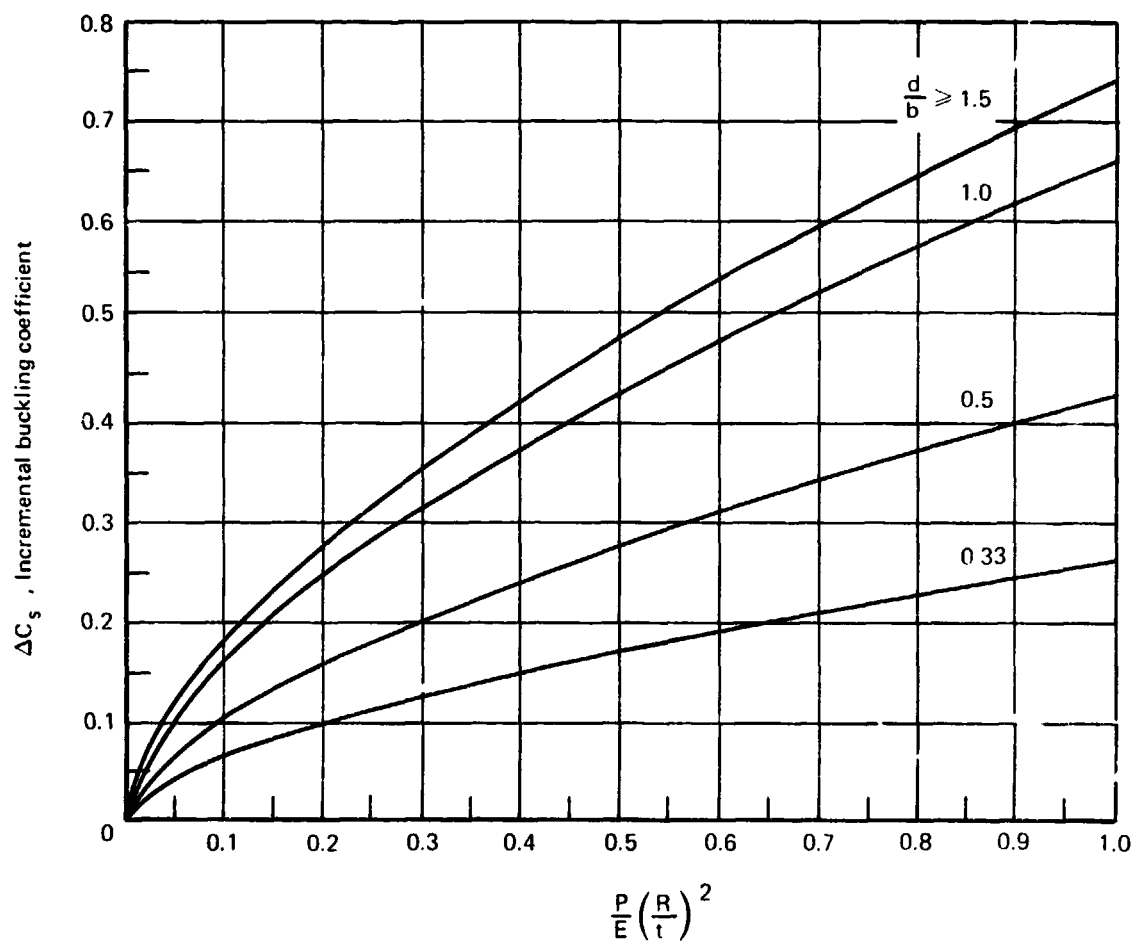


Figure 4-16.—Increase in Shear Buckling Stress of Curved Panels Due to Internal Pressure

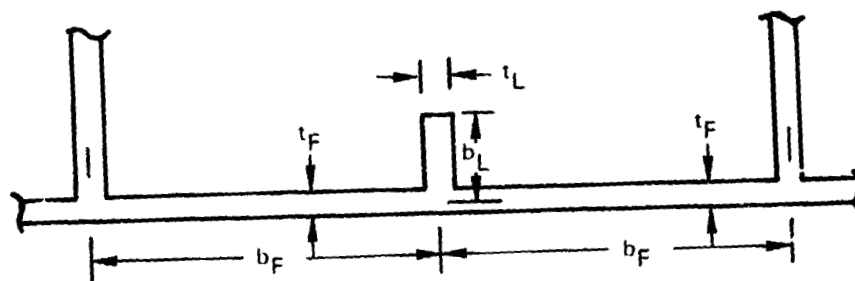
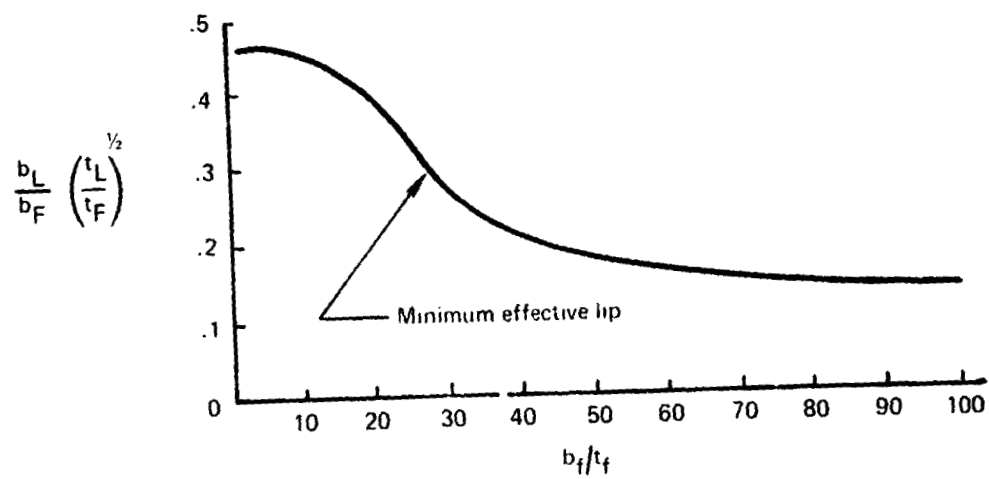


Figure 4-17.—Tab Criterion

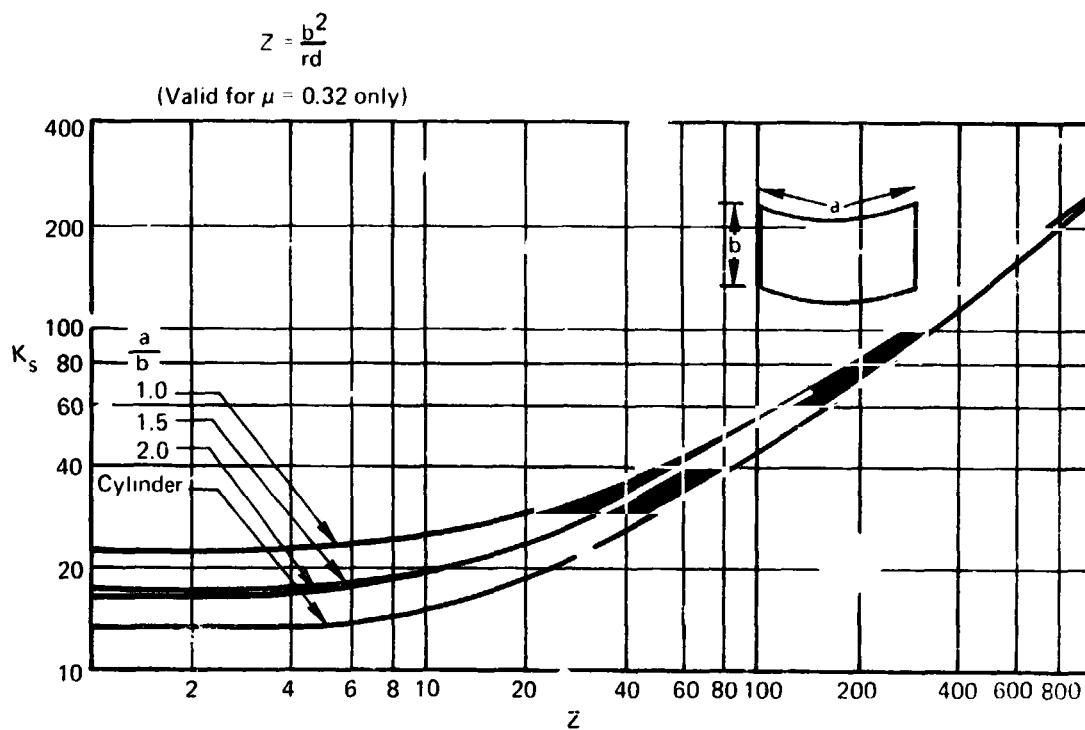
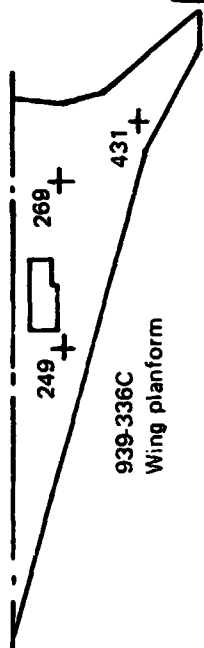


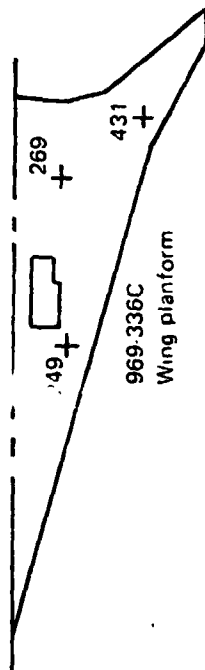
Figure 4-18.—Buckling Coefficients for Simply Supported Curved Shear Panels



Panel concept	Panel weight (lb/ft ²) without insulation					
	Pt 249		Pt 269		Pt 431	
	Upr	Lwr	Upr	Lwr	Upr	Lwr
Corr core Ti	—	—	6.038 Baseline ⁽¹⁾	—	6.575 Baseline ⁽¹⁾	4.867 (1) Baseline
Stressskin Ti H/C	1.357 Baseline ⁽¹⁾	1.464 Baseline ⁽¹⁾	—	3.466 (1) Baseline	—	—
Al brazed Ti H/C	1.521	1.626	4.921	3.813	5.331	4.957
Steel brazed H/C	—	—	7.330	5.701	6.736	6.521
Sheet stiff Ti	—	—	6.030	4.042	7.690	5.454
Integrally Mach. and welded	—	—	5.247	3.739	6.050	4.427
Integrally mach. waffle	—	—	6.392	3.772	8.382	5.035

(1) 969-336C baseline

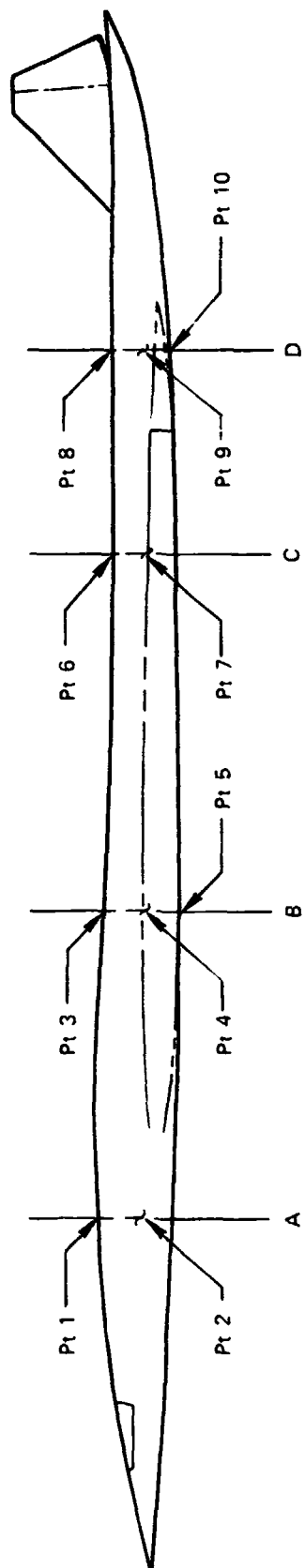
Figure 4-19.—Wing Panel Weights, 969-336C



Spar concept		Spar weights lb/in ft		
		Pt 249	Pt 269	Pt 431
Conventional*		3.349*	2.541	1.723
Sine wave*		3.086	2.043*	1.471*
Welded		3.117	2.390	1.611
Extruded		3.154	2.520	1.662
Brazed Ti		4.262	2.724	1.876
Brazed steel		6.364	4.002	2.652

*969-336C

Figure 4-20. — Wing Spar Weights, 969-336C



Panel concept	Panel weight, (lb./ft. ²)									
	Pt 1	Pt 2	Pt 3	Pt 4	Pt 5	Pt 6	Pt 7	Pt 8	Pt 9	Pt 10
Semimonocoque (baseline)	1.90	2.09	2.68	2.09	5.24	7.59	5.05	4.34	2.89	5.98
Integrally stiffened skin	2.19	2.60	3.11	3.51	4.95	8.08	6.16	4.32	4.05	5.32
Al brazed Ti H/C sandwich	1.81	1.84	2.68	2.09	4.51	7.73	4.34	3.23	2.53	5.67

Figure 4-21.— Body Shell Weights

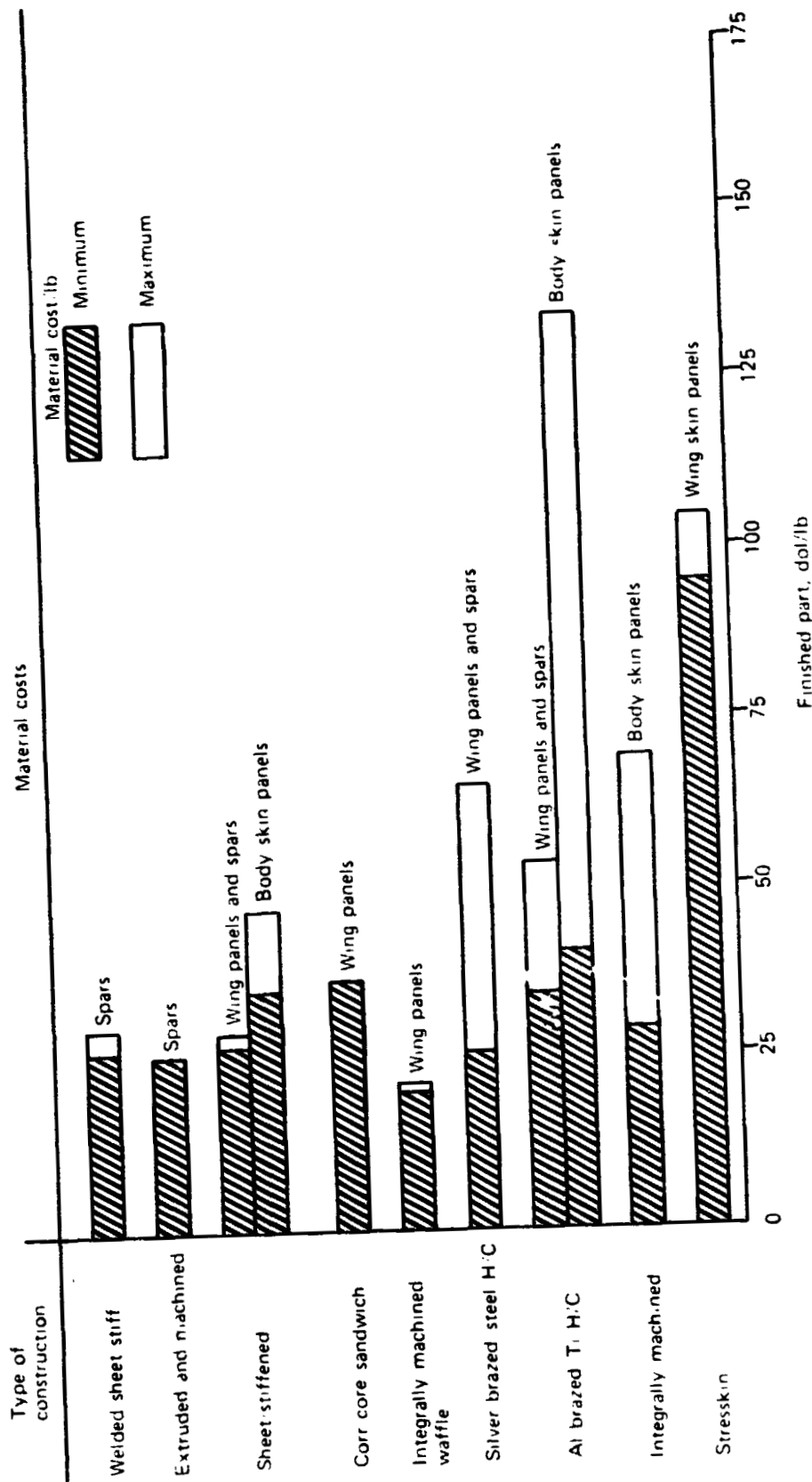


Figure 4.22. - Material Cost Analysis

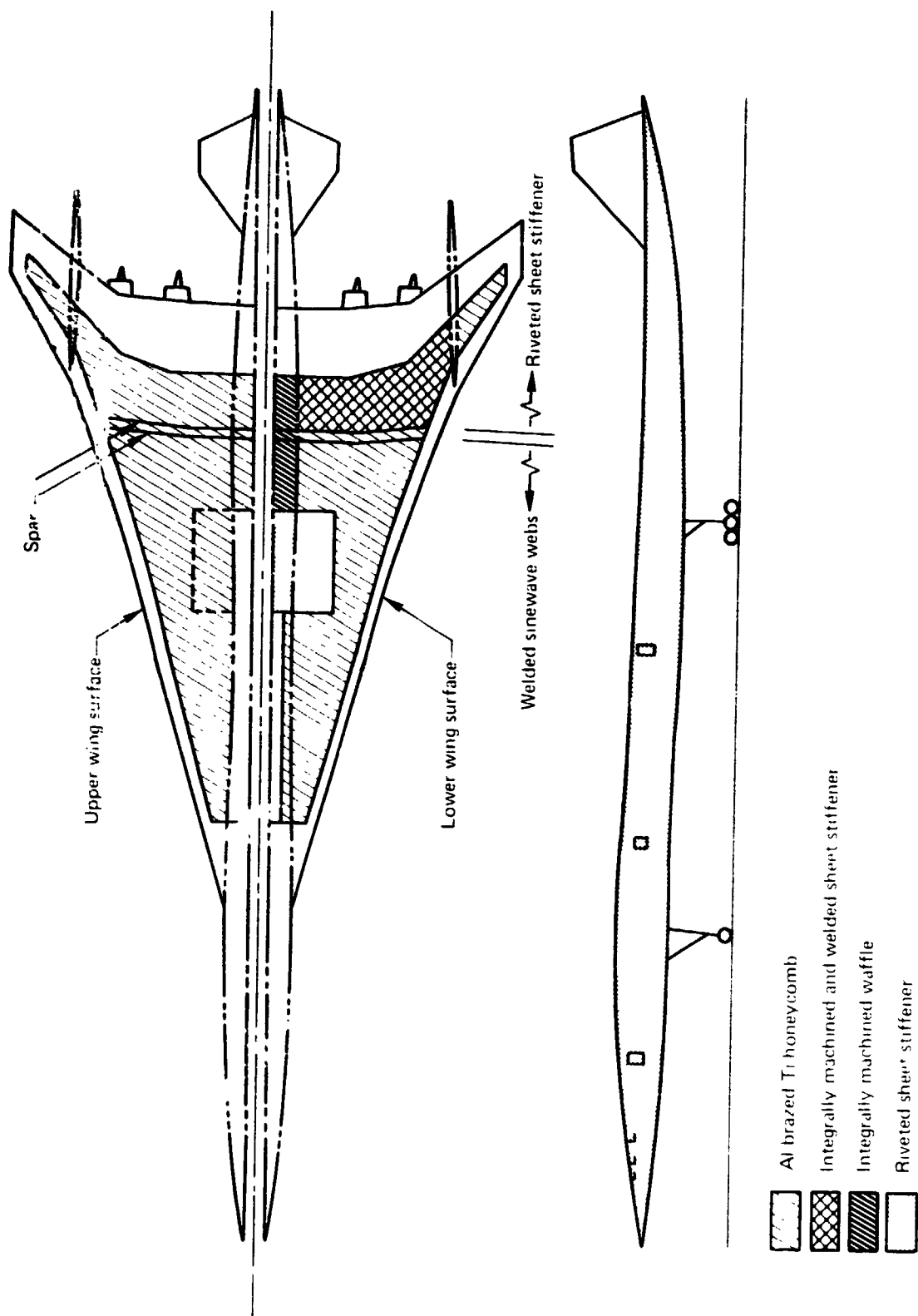


Figure 4-23 - Structural Selection, Wing and Body